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WADD TECHNICAL REPORT 60-477

Structural Test Program
F-106A Airplane

Sanford Lustig
David W. Jackson
Fred E. Hussong

ENGINEERING TEST DIVISION

AUGUST 1960

NOX

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WRIGHT AIR DEVELOPMENT DIVISION

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August 1960

System Nr. 201B
Project Nr. 1396
Task Nr. 13813

Wright Air Development Division
Air Research and Development Command
United States Air Force
Wright-Patterson Air Force Base, Ohio

FOREWORD

This report was prepared by the Wright Air Development Division as a formal record of the complete structural test program for the F-106A airplane. The structural tests reported were conducted by the Engineering Test Division, Flight and Engineering Test Group, Wright Air Development Division, Wright-Patterson Air Force Base, Ohio, with Mr. Sanford Lustig acting as Project Test Engineer; Mr. David W. Jackson responsible for the heating methods used, and Mr. Frederick E. Hussong responsible for all instrumentation.

~~This report is classified CONFIDENTIAL because the appendix presents load factor and gross weight information.~~

ABSTRACT

The F-106A airplane was subjected to a complete static test program covering all of the critical flight, landing and ground handling conditions. The F-106B was also qualified on the basis of these tests because of the structural similarity. The test loads used were the maximum loads for either the F-106A or B. The entire structure supported the required ultimate loads without modification for all conditions including conditions for which the gross weights were increased over the original design gross weights. The wing and elevon each sustained one minor local failure at a high load level. In both cases the airplane continued to support ultimate load despite these failures. Recommendations are included for structural changes necessary to eliminate the above mentioned deficiencies.

PUBLICATION REVIEW

This report has been reviewed and is approved.

FOR THE COMMANDER:

Approved by:

JOSEPH DAVIS, JR., Colonel, USAF
Commander, Flight and Engineering Test Group

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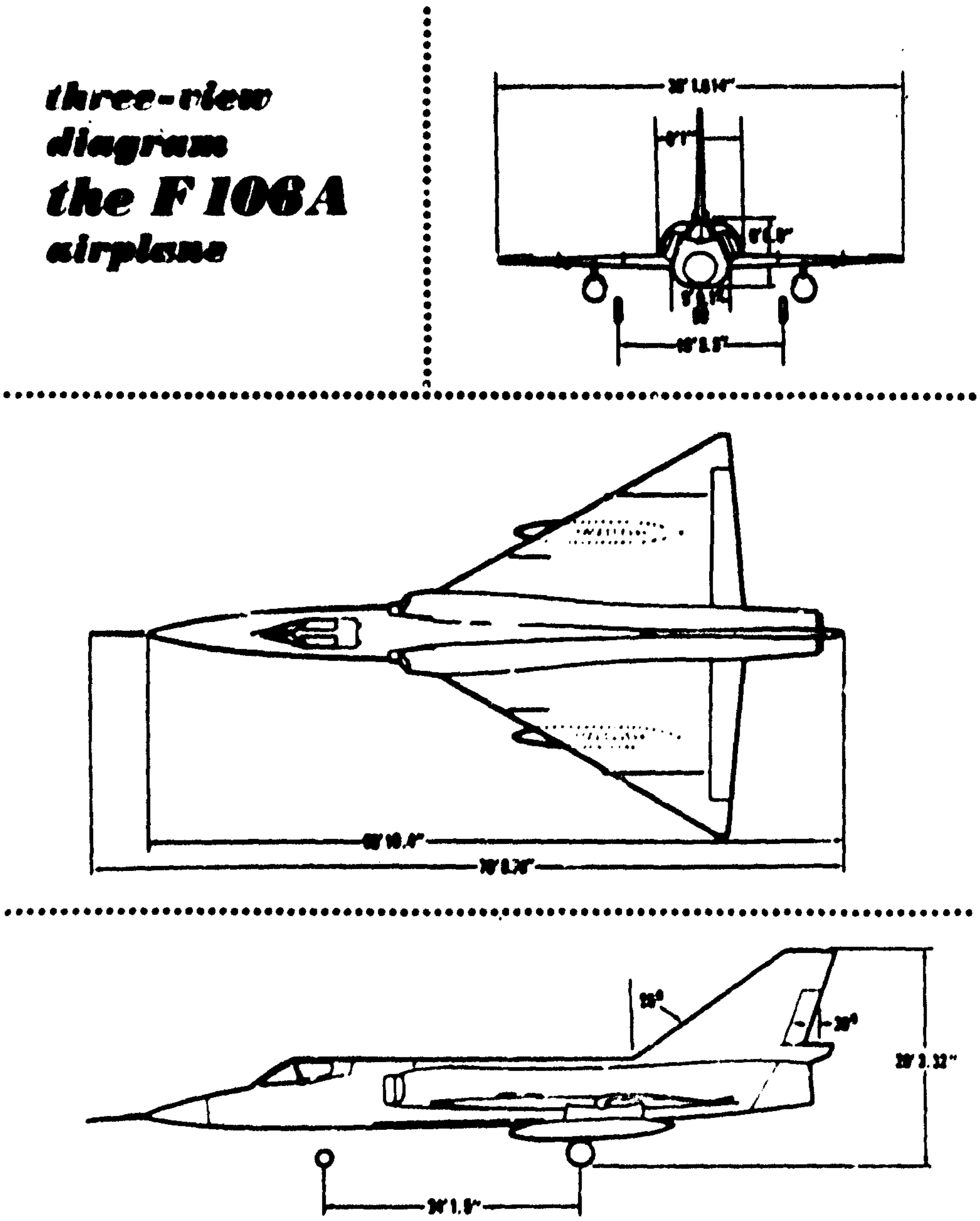

LIST OF ABBREVIATIONS AND SYMBOLS

Alt.	=	Altitude	
B. L.	=	Buttock Line	
C. G.	=	Center of Gravity	
Cond.	=	Condition	
F. S.	=	Fuselage Station	
Ft.	=	Feet	
G, g	=	Gravity Acceleration, 32.2 ft/sec^2	
G. W.	=	Gross Weight	
M	=	Mach Number	
n_x	=	Longitudinal Load Factor	
n_y	=	Lateral Load Factor	
n_z	=	Vertical Load Factor	
W. L.	=	Water Line	
$\ddot{\theta}$	=	Pitching Acceleration	Presented in radians/sec^2
$\ddot{\psi}$	=	Yawing Acceleration	
$\ddot{\phi}$	=	Rolling Acceleration	
v_x	=	Drag Load	
v_y	=	Side Load	
v_z	=	Vertical Load	

NOTE: Condition numbers followed by the letter B denote F-106B test conditions. Those numbers without any following letters denote F-106A test conditions.

INTRODUCTION

This report presents the results of the structural tests conducted on the complete airframe of the Convair F-106A airplane. These tests are of particular interest because they represent the first effort at a full scale elevated temperature structural test program. This means that aerodynamic heating of one complete wing was simulated for the temperature critical conditions, and simulated engine heat was applied throughout the entire engine compartment for all aft fuselage critical conditions. Several entirely new methods of load application were used for the first time to properly accomplish the elevated temperature tests. It was also necessary to simulate cold fuel in the wing fuel tanks to duplicate the temperature gradients required for the wing heat tests. Instrumentation requirements were satisfied by the use of elevated temperature "bakelite" strain gages and capacitance welded thermocouples.

PRELIMINARY CONSIDERATIONS

Prior to beginning the F-106 static test program, it was decided to test only the F-106A airplane and consider these tests as also representing substantiation for the F-106B. The two airplanes are structurally similar except for the cockpit area of the forward fuselage, the F-106A is a single seat and the F-106B a two seat airplane. The test loads required for any condition would be the higher of either the F-106A or F-106B. To expedite the program, it was also decided that the static test airplane would have the then available Case XIV wing which is identical structurally to the production Case XXIX wing except in the leading edge area which is structurally similar.

Actuating cylinders for such items as the armament doors and landing gear fairing doors are pneumatically operated on the F-106 aircraft. For convenience in testing, all actuating systems were converted to hydraulic operation for the static test article only. This enabled the existing hydraulic system at the WADD structural test facility to apply the proper pressures to the actuating cylinders for all conditions that required loading or reacting pressures in the cylinders.

Immediately after the decision was made to include among the F-106A static tests full scale elevated temperature tests, a method for loading the heated wing had to be decided upon. The standard method of applying load through neoprene rubber tension pads would not suffice due to the fact that the bonding materials used will not withstand temperatures much above ambient room temperature. At the time a decision had to be made, there was no known high temperature tension pad at a usable state of development. It was therefore decided to have special fittings built into the basic structure to which load could be applied directly. In this case, such an approach was relatively convenient in view of the fact that most of the F-106 wing is of standard built-up rib and spar construction tied together with standard fasteners. A more detailed description of the load fittings used and their method of attachment will follow in succeeding paragraphs.

TEST ARTICLE AND LOAD APPLICATION METHODS

The test article consisted of a complete F-106A airframe and integral pylon-tank. All major structural tests were conducted using a floating test set-up (reference typical test

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photograph, Figure 1). In this procedure the entire airframe is tested as one integral unit with the dead weight of the structure and all attached test fixtures relieved by lead weights suspended from pulleys and attached to the test article. This caused the airframe to float at 0 g's; all test loads were required to be uniformly applied and perfectly balanced in translation and roll, pitch, and yaw.

The wings and elevons were loaded primarily through fittings integrated with the basic structure. The basic wing had specific spar bolts replaced with a special bolt-stud combination fitting (reference Figure 2). The elevons, wing tips, and leading edges had tabs welded or riveted to each rib with studs screwed into the tabs and protruding through the skin (reference Figures 3 and 4). The internal tab attachments for the leading edge can be seen in Figure 5. Cables were attached to each of these loading studs and groups of cables were interconnected by means of steel or aluminum "whiffle trees". All loads were hydraulically applied. For most major conditions the load fittings in portions of the leading edges, wing tips and elevons were insufficient for the magnitude of load or were not arranged so as to be able to attain the proper center of pressure for the applied loads. This was brought about by the fact that the load fitting design had to be completed and fabrication begun before the basic loads were finalized. In such cases it was necessary to supplement the load fittings with tension patches bonded to the surfaces. Neoprene sponge rubber tension pads were used for room temperature tests. For elevated temperature tests it was necessary to use metal-to-metal tension plates bonded to the surfaces with Dow-Corning RTV Silastic. Fuselage loads were applied hydraulically through riveted or bonded shear straps and tension pads. Here again, Silastic bonded shear straps or tension pads were used for elevated temperature tests. The fin loads were applied at room temperature only and therefore loads were primarily applied through neoprene rubber tension pads; however, for conditions with simulated engine heat, the lower portion of the fin became hot enough to require Silastic tension pads. Test load application was accomplished with Edison hydraulic pressure control units and manual hydraulic control units. The manual units were primarily used for control of inherent pitch, roll, or yaw in the floating test set-up.

INSTRUMENTATION

The aircraft was instrumented by Convair-San Diego in accordance with WADD structural testing requirements. Additional strain gages and thermocouples were added at WADD during the test program. Sensing elements consisted of Baldwin-Lima-Hamilton Corporation SR-4 Bakelite type bonded wire strain gages at approximately 484 locations. Strain gages were incorporated into modified Wheatstone bridge circuits and wired for sensitivity to axial, bending and shear strains. Thermocouples were capacitance welded to the structure at all accessible locations. Junctions inaccessible for welding techniques were cemented with aluminum filled epoxy cement.

Bridge outputs were recorded by Gilmore Industries Model 114 high speed 144 channel strain gage graphical plotter. Switching was done through three modified Nosker strain indicators. Multiple passes of the chart paper through the recorder resulted in a plot of strain versus percent ultimate load. Speed of operation with this instrument is one channel per second. Sensitivity may be varied from 2000 to 20,000 micro-inches per inch full scale. Portable SR-4 strain indicators were used for manually recording outputs of 240 ohm bridges as well as monitoring compression load cells and tension straps. Thermocouples were recorded manually during steady state soak temperature conditions using a modified 84 channel Brown self-balancing pyrometer potentiometer. Control thermocouples were recorded by single channel Brown self balancing pyrometer recorders. Hot wing transient heating condition temperatures were recorded on Century Model 408 oscillographs.

A detailed description of recording instruments, transducer characteristics, method of installation, electrical wiring circuits, type of output information and transducer locations on the aircraft are on file in the WADD Structural Test Facility (WWFESS).

ELEVATED TEMPERATURE APPLICATION

Thermal loads, in addition to the mechanically induced static test loads, were introduced for those test conditions summarized in Table 1 of the Appendix. Two thermal simulations were sought in these tests, i.e., (1) the steady state conditions which were specified for the engine compartment, and (2) the transient conditions which were specified for wing heating. No attempt was made to introduce the combined effects of engine compartment and wing heating during the course of these tests, due primarily to the limited amount of power distribution equipment available for use.

Radiant heating techniques were utilized for both types of thermal simulation. The basic heating elements used were General Electric 1000T3/CL infrared heating lamps. These lamps were mounted on aluminum alloy reflector units specially fabricated and contoured to the surface being heated. Comments relative to the elevated temperature testing will pertain first to the engine compartment (or steady state) heating and secondly, to the wing (or transient) heating conditions.

Early discussions between WADD and Convair personnel led to the concept of simulating engine heat by means of a dummy engine (or can) heated from within with radiant heating elements so as to provide the required temperature distributions. After examination of the dummy engine fixture, it was concluded that the large thermal inertias involved would make control extremely difficult. This approach was therefore abandoned in favor of mounting the lamps to reflector units so arranged as to introduce the heat flux directly to the inside flange of the bulkhead frames and to the inside surfaces of the stiffened skins between the bulkheads.

To arrive at a reasonable lamp distribution for the frames, the frame cross-sectional areas, width of flange, depth of frame normal to the inside flange, and frame materials were considered. Thermocouples were mounted on 15 points on the flanges of the bulkhead located at Fuselage Station 520.0 and at 8 locations on the remaining frame stations. T-3 lamps were attached to brackets mounted from the inner flanges of the frames so that the axis of the lamps followed the contour of the flange, that is, perpendicular to the engine thrust line. Reflectors were then attached to the mounting brackets so as to reflect the radiant flux towards the frame flanges. Figure 6 portrays the arrangement of the heating elements and reflector units. Calculated distributions were good only for first approximations and actual lamp distributions depended on a "cut-and-try" technique.

Heating of the bay areas between the bulkheads was accomplished by mounting the heating lamps directly to reflector units which were contoured to hold the elements approximately four inches from the surfaces to be heated. Supporting brackets for the aluminum alloy sheet reflectors were mounted by means of bolting to fuselage fittings, utilizing numerous pilot holes as attach points. Control thermocouples for the bay areas and frames were located in areas selected symmetrically on either side of center (an unfortunate choice since compensation for conduction effects could have been better controlled by using vertical increments, i.e., control thermocouples at top and bottom).

The temperature distribution sought in the engine compartment for the applicable test conditions and the temperatures actually achieved during the tests are summarized in Table 2 of the Appendix. Details of the thermocouple locations are on file in the WADD Structural Test Facility (WWFESS). During the initial test conditions (19 through 1404-B) the maximum temperatures in the compartment were held at or below the maximum specified for the given test conditions. Several of the thermocouple readings were substantially below the desired temperature. During the final phases (Conditions 2502 and 3005), an attempt was made to bracket the required soak temperatures, except that temperatures were held to a maximum of 20°F. in excess of those required.

Wing heat tests were conducted by introducing transient heating conditions. Fuel conditions were specified for the purpose of maximizing thermally induced stresses. Both wings were identically gaged (strain and deflection) although only the left wing was subjected to heating. This instrumentation duplication was for the purpose of assisting in differentiating between thermally and mechanically induced stresses.

The wing reflectors were formed to the wing contours and supported by means of inverted hat fittings which were fastened to the contractor-installed panel point load fittings. The leading edge reflectors were bent to the required contour and were held to shape by means of aluminum alloy sheet cut to fit and clipped to the reflector by rolling the reflector edges, and by the installed baffles (intended to localize the heat flux being distributed to the selected control areas). Those reflectors over flat surfaces were stiffened by means of 1 x 1 inch "T" extruded material and by means of the spar baffles. The reflector units were fabricated in convenient sizes to faciliate installation and removal of individual reflector units as required. Lamp spacings over the wing surfaces were calculated based on equivalent skin thicknesses and calculated temperature rise rates. Fuel areas required consideration of the quantity of heat absorbed by the fuel simulant (ethelyn-glycol and water mixture). This was estimated by the contractor to be approximately 50 percent of the heat flux introduced to the wing surface. In consideration of the power available for distribution and a reasonable breakdown of control areas, it was decided to eliminate thermal loading of the elevons. The 40 control areas were distributed, 19 to the upper surface and 21 to the lower surface. Both the upper and lower wing surfaces were divided into control areas as follows: The area forward of Spar Nr. 1 was broken into two control areas, the wing tip one control area, the spars and root areas eight control areas, and the remaining wing areas taking up the remaining apportioned controllers for each of the upper and lower surfaces. Control areas and monitoring thermocouple locations may be found in detail in the WADD Structural Test Facility files (WWFESS).

The required transient wing heating conditions were programmed through the WADD heat computers. These computers, used with the saturable reactor controls, continuously compute and control the thermal input to each of the selected control areas in accordance with the following convective heat transfer and power control equations:

$$Q = h\,(T_{aw} - T_s) \qquad (1)$$

Where: Q = Rate of heat transferred (BTU/hr-ft^2)

h = Thermal convective heat transfer coefficient (BTU/hr-ft^2 in °F.)

T_{aw} = Adiabatic wall or recovery temperature (°F.)

T_s = Actual skin or surface temperature (°F.)

and

$$Q = KEI \quad (2)$$

Where: K = Multiplier which includes a series of factors peculiar to the computer-controller operation (nondimensional)

E = Line voltage

I = Line amperage

Two conditions were selected by the Contractor as representing the most severe transient thermal conditions to be encountered in actual flight: (1) a 60-degree, 21-second power dive from M = 1.3 at 60,000 feet to M = 1.895 at 30,000 feet followed by an extended cruise under the latter condition for an additional 60 seconds; and (2) a 234-second level flight acceleration from M = 1.0 to M = 2.0 at 35,000 feet followed by an additional 60-second cruise at M = 2.0.

For the purpose of these tests the wing was divided into zones as indicated in Figure 7 wherein the variance of the convective heat transfer coefficient was not over 10 percent. Computer input functions for the wing tests required time dependent thermal heat transfer (h) functions for each of the 40 selected control areas. These h functions were related to the distances aft of the leading edge of the wing as determined by control thermocouple placements. The control thermocouples provided the skin temperature feedback required for computer solutions of Equation 1. The recovery temperatures input functions for the test conditions were in the form of contractor-furnished boundary layer temperature versus time curves. Calculated boundary layer temperatures, flux requirements, thermal heat transfer coefficients, and predicted skin temperatures for the fuel and dry skin conditions are graphically portrayed in Figures 8A through 8G for the 60-degree power dive condition, and Figures 9A through 9G for the level flight acceleration condition.

Fuel simulation for the foregoing conditions was accomplished by introducing a water-ethylene glycol mixture to the tanks. The simulated fuel was precooled by means of solid carbon dioxide blocks dropped into the mixture held in an external storage container. Cooling was continued to a level several degrees below the required initial wing temperature prior to being pumped into the wing. This allowed for subsequent heat exchange between the fluid and the structure. Each of the four wing fuel tanks was independently filled for accurate fuel level control. The fuel level was of extreme importance since the fuel was not to touch the upper wing skin at any time and was to be in contact with the lower wing skin at all times. This was necessary to prevent the control thermocouples from feeding erroneous information to the computers. For example, if some cold fuel simulant was in contact with a small upper surface area that happened to contain a control thermocouple, that entire control area would be subjected to overheating because most of that area would actually be "dry" and a great deal warmer than the control thermocouple would indicate. The reverse situation would be true if the fuel simulant did not contact all lower surface control points. Independent venting of each fuel tank was necessary to prevent overpressurization of the tanks from escaping CO_2 gases from the fuel simulant.

The arbitrarily selected thermal loading conditions were superimposed upon Loading Conditions 1705 and 1407. No attempt was made to program the loads in accordance with a flight plan related to the thermal conditions imposed. Incremental loading techniques were used for both wing conditions investigated, except that differing methods were used in applying the final 10 percent load increment. For the condition where the 60-degree power dive thermal simulation was used, the mechanically induced loads were introduced incrementally up to the maximum load level desired (limit or ultimate); this load level was maintained while the entire heating cycle was introduced (80 seconds), and then the loads were incrementally reduced. For the level flight acceleration thermal simulation (300 seconds), the heating cycle was started after stabilizing at 90 percent of ultimate load. After approximately 100-seconds elapsed time of the heating cycle, the final 10 percent load increment was introduced (without interruption of the heating cycle) and held to the end of the 300-second run. At the end of the run the mechanical loads were incrementally reduced.

A detailed evaluation of the thermocouple data has not been accomplished; however, cursory examination of the data reveals a reasonable correlation with theoretical calculated results (some of which were experimentally verified under controlled conditions, i.e., water box fuel simulation). In those cases where an appreciable error appeared to exist between calculated and actual results, the apparent error could usually be attributed to: (1) recording instrument error resulting from either instrument malfunction or calibration error, (2) location of the thermocouple in an area of an uncompensated heat sink or in an area lacking a compensated heat sink (i.e., the fuel level changed somewhat in the fuel compartments due to structural deformations and translations), and/or interaction between heating areas (that is, thermocouples driven by heat flux from adjacent areas). Temperature data for both of the transient wing conditions were recorded by means of strip charts (Brown Electronik Recorders) and oscillograph recorders. This data is available at the Wright Air Development Division, (WWFESS), Wright-Patterson Air Force Base, Ohio, for review by interested and qualified requesters.

TEST CONDITIONS, DATES OF TEST, AND SUMMARY OF TEST RESULTS

The F-106A was tested for the conditions listed below:

Test Sequence	F-106A Test Condition	Test Date	Percent Ult. Load Supported
1	Canopy and Cockpit Ground Pressurization	2 December 1957	100
2	Rudder Controls Conditions 7, 8, and 9	4 December 1957	100
3	Rudder Feel System	4 December 1957	100
4	Elevator Controls System Condition 4	5 December 1957	100
5	Elevator Controls System Conditions, 1, 2, 3, & 5	6 December 1957	100
6	Elevator Feel System Conditions 1 and 3	9 December 1957	100

Test Sequence	F-106A Test Condition	Test Date	Percent Ult. Load Supported
7	Aileron Controls System Conditions 2, 3, & 4	9 December 1957	100
8	Power Controls Subsystem Conditions 1, 2, & 3	10 December 1957	100
9	Condition 1602	25 March 1958	100
10	Condition 1610	2 April 1958	100
11	Condition 1604	10 April 1958	100
12	Condition 1704	24 April 1958	100
13	Condition 5	21 May 1958	100
14	Condition 15	5 June 1958	100
15	Falcon Launcher Condition 1 (Retracted)	26 June 1958	100
16	Drag Chute (at 18 Degrees)	27 June 1958	100
17	Drag Chute (at -5 Degrees)	30 June 1958	100
18	Ram Air Turbine Door Condition 1-C	3 July 1958	100
19	Ram Air Turbine Door Condition 3	3 July 1958	100
20	Condition 19 (With Engine Heat)	10 July 1958	100
21	Condition 2 (With Engine Heat)	24 July 1958	100
22	Condition 19 - F-106B (With Engine Heat)	31 July 1958	100
23	Condition 1904	12 August 1958	100(97)
24	Condition 1806 (With Engine Heat)	20 August 1958	100
25	Condition 1902	26 August 1958	100
26	Condition 1404 - F-106B (With Engine Heat)	3 September 1958	100
27	Armament Doors Condition 2	2 October 1958	100
28	Armament Doors Condition 8	10 October 1958	100
29	Armament Doors Condition 13C	16 October 1958	100
30	Armament Doors Condition 14C	17 October 1958	100
31	Speed Brakes - 50 Degrees Open	21 October 1958	100

Test Sequence	F-106A Test Condition	Test Date	Percent Ult. Load Supported
32	ECP 4056 Controls - Rudder Condition 5	[illegible] November 1958	100
33	ECP 4056 Controls - Rudder Condition 7	7 November 1958	100
34	ECP 4056 Controls - Rudder Condition 9	7 November 1958	100
35	ECP 4056 Controls - Brake Condition 5	7 November 1958	100
36	ECP 4056 Controls - Elevator Condition 3	12 November 1958	100
37	ECP 4056 Controls - Elevator Condition 4	12 November 1958	100
38	ECP 4056 Controls - Elevator Condition 2	13 November 1958	100
39	ECP 4056 Controls - Elevator Condition 5	13 November 1958	100
40	ECP 4056 Controls - Aileron Condition 2	14 November 1958	100
41	ECP 4056 Controls - Aileron Condition 3	14 November 1958	100
42	ECP 4056 Controls - Aileron Condition 4	14 November 1958	100
43	Main Landing Gear Wing Fairing Door Condition 6	24 November 1958	100
44	Main Landing Gear Wing Fairing Door Condition 7B	25 November 1958	100
45	Nose Landing Gear - Three-Wheel Level Landing	26 November 1958	100
46	Nose Landing Gear - Spin Up	28 November 1958	100
47	Nose Landing Gear - Spring Back	1 December 1958	100
48	Main Landing Gear Doors - Closed - Wing And Fuselage	2 December 1958	100
49	Nose Landing Gear Door - Closed	3 December 1958	100
50	Pilot Seat - Downward Crash	4 December 1958	100
51	GAR Launcher - Retracted	4 December 1958	100
52	GAR Launcher - Crash	5 December 1958	100
53	Condition 1705 (With Wing Heat)	12 December 1958	100
54	Condition 1407 (With Wing Heat)	17 December 1958	100

Test Sequence	F-106A Test Condition	Test Date	Percent Ult. Load Supported
55	Nose Landing Gear - Towing Aft	31 December 1958	100
56	Nose Landing Gear - Towing Forward	5 January 1959	100
57	Nose Landing Gear - Unsymmetrical Braking	6 January 1959	100
58	Main Landing Gear - Taxi	8 January 1959	100
59	Main Landing Gear - Side Drift Outboard	12 January 1959	100
60	Main Landing Gear - Side Drift Inboard	12 January 1959	100
61	Main Landing Gear - Side Drift with Spring-Back	14 January 1959	100
62	Nose Landing Gear - Towing 45 Degrees Aft	16 January 1959	100
63	Main Landing Gear - Two-Wheel Spin-Up	20 January 1959	100
64	Main Landing Gear - Two-Wheel Spin-Up (Tail Down)	21 January 1959	100
65	Main Landing Gear - Two-Wheel Spin-Up (Tail Down Side Load)	21 January 1959	100
66	Main Landing Gear - Two-Wheel Spring-Back (Tail Down)	23 January 1959	100
67	Main Landing Gear - Two-Wheel Spring-Back (Tail Down Side Load)	23 January 1959	100
68	Main Landing Gear - Braked Roll	26 January 1959	100
69	Main Landing Gear - Turning	27 January 1959	100
70	Main Landing Gear - Pivoting	27 January 1959	100
71	Main Landing Gear - Side Drift with Spin-Up	28 January 1959	100
72	Main Landing Gear - Mooring Fitting	28 January 1959	100
73	Main Landing Gear - Jacking	29 January 1959	100
74	Condition 2502 (I.F.)	6 February 1959	100
75	Condition 3202 (I.F.)	12 February 1959	100
76	Condition 3005 (I.F.)	19 February 1959	100

Test Sequence	F-106A Test Condition	Test Date	Percent Ult. Load Supported
77	Condition 4 (I.F.)	20 February 1959	100
78	MLG Fuselage Fairing Door - Condition 3	26 February 1959	100
79	NLG Door - Open and Locked	26 February 1959	100
80	Pilot Seat - Forward Crash	26 February 1959	100
81	Pilot Seat - Side Crash	26 February 1959	100
82	Pilot Seat - Catapult Load	2 March 1959	100
83	Pilot Seat - Forward Crash (32g)	3 March 1959	100
84	Falcon Launcher - Condition 7	3 March 1959	100
85	Falcon Launcher - Condition 7A	3 March 1959	100
86	Falcon Launcher - Condition 9	4 March 1959	100
87	Falcon Launcher - Condition 6	4 March 1959	100
88	Forward Engine Mount - Condition 2F	5 March 1959	100
89	Forward Engine Mount - Condition 5D	5 March 1959	100
90	Forward Engine Mount - Condition 19F	6 March 1959	100
91	Forward Engine Mount - Emergency Landing	9 March 1959	100
92	Forward Engine Mount - Condition 5E	9 March 1959	100
93	Forward Engine Mount - Condition 5C	10 March 1959	100
94	Aft Engine Mount - Condition 1910C	11 March 1959	100
95	Aft Engine Mount - Condition 1804C	12 March 1959	100
96	Towing Ring - Towing Condition	12 March 1959	100
97	MLG Drag Strut Lug - Power Run-Up	13 March 1959	100
98	MB-1 Ejection	18 March 1959	100
99	Hoisting - Forward Hoist Points	19 March 1959	100
100	Hoisting - Aft Hoist Points	20 March 1959	100
101	Jacking - Forward Jack Point	23 March 1959	100

Test Sequence	F-106A Test Condition	Test Date	Percent Ult. Load Supported
102	MB-1 - Forward Crash	24 March 1959	100
103	Pylon and Tank - Condition 8	24 March 1959	100
104	Pylon and Tank - Condition 12	25 March 1959	100
105	Pylon and Tank - Condition 9	25 March 1959	100
106	Pylon and Tank - Condition 1504	26 March 1959	100
107	Main Landing Gear - Condition 1102B	3 April 1959	100
108	Jacking - Wing Fitting	7 April 1959	100
109	Fixed Inlet Ramp - Condition 7	21 April 1959	100
110	Fixed Inlet Ramp - Condition 1	23 April 1959	100
111	Variable Inlet Ramp - Condition 8	29 April 1959	100
112	Variable Inlet Ramp - Condition 7	1 May 1959	100
113	Variable Inlet Ramp - Condition 11(+)	4 May 1959	100
114	Ramp Forward Actuators - Condition 11F(-)	5 May 1959	100
115	Variable Inlet Ramp - Condition 11 (-)	6 May 1959	100
116	Ramp Aft Actuators - Condition 11A (-)	7 May 1959	100
117	Ramp Aft Actuators - Condition 7A	14 May 1959	100
118	Inlet Duct Pressurization	21 May 1959	100 (Approx.)

NOTE: A detailed description of the conditions listed above appears in the appendix.

At the conclusion of the limit load portion of the Condition 1602 Test, two skin gap problems were noted. The skin gap at the aft end of the missile bay and the gap around the fuselage main landing gear doors were found to be insufficient, with resulting skin jamming. It was recommended that new skin gap tolerances be established for these areas, with the existing maximum allowable gap established as the new minimum gap.

Three attempts were made to complete Condition 1704. In each case the test had to be discontinued at as low a point as 50 percent ultimate load because of jamming of the inboard edge of the elevons against the fuselage (reference Figure 10). In each case the overhanging skin of the inboard elevon rib flange was shaved in an attempt to gain the proper clearance. This shaving was continued until it was under the proper edge distance for the inboard row of elevon rivets. At this point the Contractor advised locating another row of rivets spaced between the existing rivets and the outboard rib web approximately .25 inch outboard of the rib web. This permitted shaving the rib flange and skin to the original line of rivets. The test was conducted a fourth time and the structure satisfactorily supported 100 percent ultimate load with sufficient clearance existing at all times. Immediately after the test the Contractor advised WADD that all F-106 aircraft construction would be similar to the static article, i.e., the inboard elevon rib flange and skin would be ground down to the clearances required during the static test. This requirement was called out in Convair Drawing Nr. 8-13380.

At some point above 90 percent, ultimate load for Condition 2502, the shear-carrying elevon slip joint separated at the elevon trailing edge. The elevon continued to support load and 100 percent ultimate load was attained with no failures at any point. While reducing the load, the slip joint that had separated butted at the trailing edge separation point instead of slipping back into place. This caused skin cracking at the butting area (reference Figure 11). Elevon chordwise bending was determined to be the prime cause of the separation and correction of it was investigated and found to be difficult. In view of the high load level at the time of separation and the fact that load continued to be supported, it was agreed that no corrective action would be required at this time.

At 95 percent ultimate load for Condition 3202, a sharp compression buckle in the wing upper surface skin caused rolling of the rib cap of the B.L. 99.94 rib between Spars 6 and 7 (reference Figure 12). The rolling caused the rib cap web to crack immediately below, and sometimes through, the rib cap flange-to-web fillet radius. The structure continued to support load and the test was continued to 100 percent ultimate load without further failure. The above mentioned crack appeared between lightening holes drilled very close to the rib cap flange (reference Figure 13). In some cases the hole actually cut into the flange-web fillet radius. The holes were located in this manner for use as a lower surface rib cap fuel flow passage; the upper cap was similar because of symmetry and/or cost reduction purposes. It was recommended that these holes be moved down from the cap fillet on future production airplanes and that the possibility of fatigue problems in the existing configuration be investigated. The recommended production change was immediately implemented and details of this change may be found in Convair Drawing Nr. COR-8-00139.

Service problems with the F-102 landing gear caused concern for the similarly designed F-106 landing gear, and the possibility of a future requirement for an increased strength landing gear for the F-106. Before a redesigned gear could be installed, we must know the strength level of the gear supporting structure in the wing. It was therefore decided to conduct a destruction test for the landing condition that produced the most critical wing loads. For this particular test, the Contractor was to fabricate an overstrength dummy landing gear with which to introduce the loads. While the dummy gear was being designed, the length of time involved in its design and fabrication prompted a decision to conduct the test with the actual landing gear. This was based on the possibility that a wing failure could occur before a gear failure and thereby eliminate the necessity for the dummy gear. Condition 1102B, a two-wheel tail down spring-back condition, was selected for test because it produced very close to the maximum wing spar loads without overloading the very critical landing gear side brace boss. At 135 percent ultimate load, the landing gear forward drag strut failed in compression causing a number of secondary failures (reference Figures 14A through 14E). At this high load level the wing was still in excellent condition. The very high strength level thereby demonstrated by the F-106 wing made it unnecessary to conduct further wing tests at that time.

CONCLUSIONS

It is concluded that the F-106A and B airplanes, with the modifications noted in the Summary of Results, are structurally capable of withstanding the static ultimate loads shown in the appendix. These loads include both the original and later increased design gross weights as set forth in the appendix and also include all applicable temperature considerations.

APPENDIX

F-106 STRUCTURAL TEST CONDITIONS

(All parameters shown are limit conditions)

TABLE 1

THERMAL CONDITIONS

NR	DATE	TEST CONDITION	THERMAL CONDITION
1	10 July 58	19	
2	24 July 58	2	
3	31 July 58	19-B	Engine Compartment heat
4	20 Aug 58	1806	
5	3 Sept 58	1404B	
6	11 Dec 58	1705 Limit	60° P. Dive-Wing heat
7	12 Dec 58	1705 Ultimate	60° P. Dive-Wing heat
8	17 Dec 58	1407 Limit	Level flight acceleration, wing heat
9	17 Dec 58	1407 Ultimate	Level flight acceleration, wing heat
10	6 Feb 59	2502	Engine Compartment heat
11	19 Feb 59	3005	Engine Compartment heat

TABLE 2

SUMMARY OF FINAL (SOAK) ENGINE COMPARTMENT TEMPERATURE (F°)

BAY OR FRAME	T/C Nr.	COND. REQ.	19 17 JUL	2 21 JUL	2 23 JUL	2 24 JUL	198 31 JUL	REQ.	1806 20 AUG	1404B 2 SEP	1404B 3 SEP	REQ	2502 6 FEB	3005 18 FEB
1	1	190	121		131	132	137	200	146	139	154	190	142	122
	1-S	165	165	165	165	165	165	185	185	185	200	178	205	205
	2	185	157		168	154	159	195	184	179	189	185	172	176
	2-S	170	170	170	170	170	170	190	190	190	200	177	190	190
	3	185	153		152	148	151	195	170	168	179	185	170	170
	3-S	170	170	170	170	170	170	190	190	190	205	177	183	180
	4	190	190		OUT	OUT	OUT	200	OUT	OUT	OUT	190		
1	4-S	165	OUT	165	165	165	165	185	200	200	200	173	190	190
520.0	1	230	237		240	240	240	240	250	248	253	248	264	258
	2		OUT											
	3		226		233	235	235		241	239	[illegible]		256	245
	4		OUT											
	5		226		230	232	233		246	238	240		250	240
	6		230	230	230	230	230		240	240	240		250	250
	7		229		226	229	226		238	232	232		263	259
	8		OUT											
	9	230	217		209	210	208	240	216	215	214	248	245	236
	10	195	OUT					200				193		
	11													
	12													
	13													
	14													
520.0	15		OUT											
2	5		195	195	195	195	195		200	200	205		200	200
	5-S		172		174	[illegible]	180		180	180	185		177	175
	6		195	195	195	195	195		200	200	205		195	195
	6-S		178		181	179	183		192	190	197		179	175
	7		195	195	195	195	195		200	200	205		200	200
2	7-S	195	179		174	174	177	200	185	178	182	193	179	210
556.75	1	230	230	230	230	230	230	240	240	240	250	248	265	265
	2		217		215	216	215		224	227	235		244	248
	3		211		202	205	203		210	217	217		246	244
	4		206		201	204	202		212	213	216		236	228
	5		201		192	195	194		208	205	207		234	228
	6		200		191	194	192		206	205	206		234	230
	7		197		187	190	186		196	199	201		222	222
556.75	8	230	195		187	190	188	240	194	199	200	248	220	218
3	8	195	195	195	195	195	195	200	200	200	215	193	210	210
3	8-S	195	161		150	154	152	200	164	159	168	193	175	210
569.4	1	230	230	230	230	230	230	240	240	240	250	248	265	265
	2		217		213	215	212		230	226	239		247	242
	3		200		200	203	200		218	220	222		230	220
	4		194		190	194	191		212	205	218		228	216
	5		207		200	205	197		224	217	216		246	238
569.4	6	230	196		189	192	188	240	210	207	207	248	234	236

(TABLE 2 CONT.)

BAY OR FRAME	T/C Nr.	COND. REQ.	19 17 JUL	2 21 JUL	2 23 JUL	2 24 JUL	198 31 JUL	REQ.	1806 20 AUG	1404B 2 SEP	1404B 3 SEP	REQ.	2502 6 FEB	3005 18 FEB
569.4	7	230	186		177	180	174	240	195	193	196	248	218	202
569.4	8	230	195		189	196	187	240	215	212	217	248	243	239
4	9	195	195	195	195	195	195	200	200	200	215	193	210	210
	9-S		164		156	155	156		168	162	170		171	155
	10		195	195	195	195	195		200	200	215		210	210
4	10-S	195	152		155	155	155	200	168	156	167	193	164	164
593.46	1	230	228		222	221	218	240	236	232	243	248	256	256
	2		230	230	230	230	230		240	240	250		265	265
	3		220		216	216	212		226	224	235		250	243
	4		201		198	198	195		210	207	214		232	227
	5		OUT										218	220
	6		194		190	190	186		200	195	195		214	222
	7		200		196	193	191		200	200	204		210	222
593.46	8	230	200		196	193	191	240	200	199	205	248	209	217
5	11	195	195	195	195	195	195	200	200	200	215	193	210	210
	11-S		165		154	158	155		178	171	185		217	208
	12		186		181	180	181		194	185	192		194	189
5	12-S	195	195	195	195	195	195	200	200	200	205	193	200	200
614.8	1	230	212		213	214	214	240	226	225	233	248	250	246
	2		230	230	230	230	230		240	240	250		265	265
	3		229		225	224	223		238	236	250		258	254
	4		230	230	230	230	230		240	240	250		255	255
	5		212		211	208	206		220	220	226		230	236
	6		192		190	185	183		198	198	201		204	212
	7		185		180	185	181		188	188	191		184	192
614.8	8	230	192		186	184	181	240	194	192	200	248	194	196
6	13	195	209	195	195	195	195	200	200	200	215	193	179	157
	13-S		160		164	164	165		172	164	175		210	210
	14		200	195	195	195	195		200	200	215		179	179
6	14-S	195	175		167	166	167	200	178	170	180	193	210	210
665.38	1	230	218		211	213	213	240	230	228	233	248	270	270
	2		220		219	217	218		230	235	243		275	270
	3		230	230	230	230	230		240	240	250		265	265
	4		214		213	213	213		224	228	240		254	255
	5		198		194	193	192		204	211	215		243	253
	6		178		155	157	160		174	173	177		199	218
	7		191		164	166	170		178	179	193		197	213
665.38	8	230	200		187	189	191	240	180	182	194	248	202	216
7	15	195	195	195	195	195	195	200	200	200	215	193	200	200
	15-S		169		177	170	178		172	174	201		152	156
	16		195	195	195	195	195		200	200	215		200	200
7	16-S	195	160		158	158	163	200	180	169	197	193	154	152
672.38	1	255	235		227	231	231	260	244	240	250	257	265	261
672.0	2	255	236		233	237	236	260	250	250	262	257	274	276
668.5	3	245	245	245	245	245	245	250	250	250	260	253	270	270

(TABLE 2 CONT.)

BAY OR FRAME	T/C Nr.	COND. REQ.	19 17 JUL	2 21 JUL	2 23 JUL	2 24 JUL	19B 31 JUL	REQ.	1806 20 AUG	1404B 2 SEP	1404B 3 SEP	REQ.	2502 6 FEB	3005 18 FEB
663.75	4	235	214		211	213	214	240	210	204	223	250	216	212
660.0	5	230	206		192	196	198		210	208	221	248	236	248
	6		187		160	169	175		174	164	177		190	210
	7		207		189	191	196		188	186	206		202	212
660.0	8	230	209		197	197	199	240	189	191	211	248	202	205
8	17	225	225	225	225	225	225	210	210	210	220	207	220	220
8	17-S	255	235		223	223	223	235	204	199	207	228	206	206
9	18	235	209		209	201	211	215	OUT	199	199	210	198	194
	18-S	215	215	215	215	215	215	205	205	205	205	200	210	210
	19	275	275	275	275	275	275	245	245	245	245	240	250	250
	19-S	270	275		226	258	299	245	250	204	230	238	250	248
	20	295	278		OUT	OUT	OUT	270	OUT	OUT	OUT	265	244	236
	20-S	295	295	295	295	295	295	270	270	270	270	265	265	265
	21	340	244		290	279	275	330	266	268	275	325	275	273
9	21-S	320	340	340	340	340	340	305	305	305	320	297	310	310
	1-T		132											
	2-T		169											
	3-T		117											
	4-T		104											
	5-T		138											
	6-T		110											
400	L		94											
400	U		102											
425	L		97											
425	U		113											
460	L		123											
460	U		148											
COMM.	1		182		176	174	178		186	182	184		180	182
	2		199		199	198	198		210	207	210		206	202
	3		162		153	154	155		168	159	160		172	172
	4		186		186	184	186		196	194	201		194	190
	5		151		143	144	144		160	150	151		168	171
	6		177		179	173	171		184	167	188		188	188
	7		164		167	163	170		167	168	190		OUT	OUT
	8		183		180	180	180		192	189	207		193	192
	9		264		255	252	255		238	230	230		234	232
	10		275		253	250	291		257	208	220		250	248
	11		177		178	177	180		188	183	190		190	181
	12		188		187	187	187		196	194	201		202	198
	13		241		245	245	243		254	253	257		268	262
	14		232		235	234	232		243	240	245		253	248
	15		218		213	214	214		228	221	225		232	228
	16		214		213	215	208		OUT	OUT	OUT		226	222
	17		22[illegible]		218	218	216		232	230	241		254	253
COMM.	18		230		223	224	221		238	235	247		260	261

TABLE 3

TEST CONDITIONS (F-106*)

	TEST CONDITION	n_y	n_z	GROSS WT (lbs)	Altitude (ft)	MACH	$\ddot{\theta}$ radians per sec^2	$\ddot{\psi}$ radians per sec^2	$\ddot{\phi}$ radians per sec^2	$\ddot{\phi}$ radians per sec^2
1602	Steady state pull-up; dive brakes; no thrust.		7.0	29,776	7000	1.23				
1610	Steady state pull-up; dive brakes, no thrust		7.0	29,776	0	.80				
1604	Accelerated pull-up; dive brakes; no thrust.		7.0	29,776	28,000	1.10	-4.82			
1704	Steady state pull-up; dive brakes; no thrust.		7.0	29,776	33,000	1.64				
1407	Steady state pull-up.		5.33	23,988	41,000	2.0				
1705	Steady state pull-up.		5.33	29,766	30,000	1.895				
2502	Steady state pull-up; dive brakes; no thrust		7.0	30,590	9,000	1.23				
3202	Steady state pull-up; dive brakes; no thrust		7.0	33,119	8,000	1.20				
1904	Steady state push-over; dive brakes; no thrust		-3.0	33,000	35,332	1.755				
1806	Steady state push-over; no dive brakes; thrust.		-2.3	28,421	0	1.138				
1902	Steady state push-over; no dive brakes, thrust		-3.0	33,000	0	1.05				
1404	Steady state push-over; no dive brakes; thrust		-1.8	28,755	0	1.138				
3005	Steady state push-over, no dive brakes; thrust		-2.3	29,235	0	1.138				
5	Bank to bank roll; no dive brakes; thrust	-1.09	5.0	28,421	0	.80	212	1.224	1.604	1.604
15	Bank to bank roll; no dive brakes, thrust	.251	3.90	28,421	0	1.138	-.110		-1.133	-1.133
2	Zero g Roll; no dive brake, thrust	-1.057	0.16	28,421	0	1.00	-0.135	1.547	8.076	8.076
19	Lateral gust; no dive brakes, thrust	-1.387	1.00	25,600	0	1.05	-1.615		7.647	7.647
19B	Lateral gust, dive brakes	1.155	1.0	30,221	0	1.05		1.378	-6.626	-6.626

* B denotes F-106B, all other conditions pertain to the F-106A

TABLE 3

TEST CONDITIONS (F-106*)

n_z	GROSS WT. (lbs)	Altitude (ft)	MACH	$\ddot{\theta}$ radians per sec^2	$\ddot{\psi}$ radians per sec^2	$\ddot{\phi}$ radians per sec^2	CRITICAL AREAS
7.0	29,776	7000	1.23				Wing tips in positive shear and bending; wing spars 3, 4, 5, 6, and 7 in positive bending; Fuselage Stations 102-140, 300-355, 472-593 in vertical shear; Fuselage Stations 355-520 in negative bending.
7.0	29,776	0	80				Fuselage Stations 160-280 in vertical shear; Fuselage Stations 120-316 in negative bending; wing spars 2, 3, and 4 in positive bending
7.0	29,776	28,000	1.10	-4.82			Fuselage Stations 80 & fwd in positive bending; wing spars 6 and 7 in positive bending.
7.0	29,776	33,000	1.64				Wing spars 3, 4, 5, 6, & 7 in positive bending.
5.33	23,988	41,000	2.0				Wing "hot" condition.
5.33	29,766	30,000	1.895				Wing "hot" condition.
7.0	30,590	9,000	1.23				Complete wing and fuselage aft of Fuselage Stations 355.
7.0	33,119	8,000	1.20				Wing and fwd fuselage
-3.0	33,000	35,332	1.755				Wing spar 6 in negative bending
-2.3	28,421	0	1.138				Fuselage in vertical shear, Fuselage Stations 499-575, 615-660; wing spar 5 in negative bending
-3.0	33,000	0	1.05				Fuselage in positive bending Fuselage Stations 472-478; wing spar 4, 5, 6, & 7 in negative bending.
-1.8	28,755	0	1.138				Fuselage in positive bending, Fuselage Station 472-620
-2.3	29,235	0	1.138				Aft fuselage in positive bending
5.0	28,421	0	.80	.212	1.224	1.604	Vertical fin bending
3.9[illegible]	28,421	0	1.138	-.110		-1.133	Maximum rudder hinge moment.
0.16	28,421	0	1.00	-0.135	1.547	8.076	Aft fuselage in torsion, fin bending; wing spar 5 outboard negative bending.
1.00	25,600	0	1.05	-1.615		7.647	Fuselage side bending; vertical fin bending.
1.0	30,221	0	1.05		1.378	-6.626	Fuselage side bending; vertical fin bending

...ions pertain to the F-106A.

TABLE 4

COMPONENTS TESTS (F-106B)
(Main Landing Gear)

Test Condition	n_z	Gross Wt. (lbs)	Tail Position	Oleo Position	Vertical Load Vz/Gear (lbs)	Drag Load V_x/Gear (lbs)	Side Load Vy/Gear (lbs)
Taxi	2.0	39,505	-	Static	35,985	0	0
Side drift	1.8	27,564	-	Fully extended	11,074	0	8,859 inboard
Side drift	1.8	27,564	-	Fully extended	11,074	0	6,644 outboard
Side drift	1.8	27,564	-	Fully extended -4.7 in.	11,074	9,7[illegible]2 forward	9,689 outboard
Side drift	1.8	27,564	-	Fully extended -4.0 in.	11,074	8,526 aft	8,829 inboard
Two-wheel spin-up	2.6	27,564	Level	Fully extended -2.0 in.	22,159	17,062 aft	0
Two-wheel spin-up	2.89	27,564	Down	Fully extended -2.0 in.	24,798	6,943 aft	0
Two-wheel spin-up	2.89	27,564	Down	Fully extended -6.0 in.	24,798	6,[illegible] aft	6,555
Two-wheel spring-back	2.60	27,564	Down	Fully extended -2 in.	21,134	-18,859	
Two-wheel spring-back	2.6	27,564	Down	Fully extended -7 in.	21,134	-18,859	4,183
Two-wheel braked roll	1.0	39,505	-	Static	29,630	23,703	-
Ground turning	1.0	39,505	-	Static	25,945/ 10,057	-	12,972/5,029
Pivoting	1.0	39,505	-	Static	18,000	Torque = 50,835 in. lbs.	
Jacking	1.35	39,505	-	Static	24,300	7,200	-7,200
Local Fittings							
							Gear Load
Mooring ring @45°	-	-	-	-	-	-	11,500
Towing ring	-	-	-	-	-	-	7,370
Power run-up	-	-	-	-	-	-	21,000

TABLE 5

COMPONENTS TESTS (F-106B)
(Nose Landing Gear)

Test Condition	n_z	Gross Wt. (lbs)	Oleo Position	V_z (lbs)	V_x (lbs)	V_y (lbs)
Three-wheel max. strut reaction	2.6	27,564	Fully extended -1.0 in.	11,782	2,945	-
Three-wheel max. spin-up	2.56	27,564	Fully extended -1.0 in.	6,458	4,972	-
Three-wheel max. spring-back	2.56	27,564	Fully extended -1.0 in.	6,458	-4,440	-
Unsymmetrical braking	1.0	39,505	Static	6,886	-	4,184
Towing aft	1.0	39,505	Static	6,822	9,265	-
Towing forward	1.0	39,505	Static	6,822	-9,265	-
Towing at 45° aft	1.0	39,505	Static	6,822	3,275	3,275

TABLE 6

COMPONENTS TESTS

(ARMAMENT LAUNCHING GEAR)

	TEST CONDITION	n_x	n_z	LAUNCHING GEAR POSITION
1	Max. vertical inertia (falcon)		7.0	Retracted
1	Forward crash (falcon)	5.33 (ultimate)*		Retracted
6	2 missiles on crossbridge, fired, and about to leave launch rails (falcon)			Down
7A	2 missiles on crossbridge, no missile thrust (falcon)			Down
7	2 missiles on crossbridge, just fired, with thrust (falcon)			Down
9	Forward installation, righthand missile only, just fired (falcon)			Down
	Ejection loads (MB-1)			
	Forward crash (MB-1)	5.33 (ultimate)*		

* Ultimate loads are compression (-), tension (+).

TABLE 7

MISCELLANEOUS COMPONENTS TESTS (F-106*) (ULTIMATE)

TEST CONDITION		n_z	Gross Wt. (lbs)	Oleo Position	v_z (lbs)	v_x (lbs)	v_y (lbs)	Door Position	Air Loads	Inertia Loads
	Nose jack, engine out	2.0	39,505	-	14,077	4,880	3,519			
	Wing jack, engine in	2.0	39,505	-	30,095	5,958	7,523			
	Fuselage, aft hoist	2.0	39,505	-	26,703					
	Fuselage, forward hoist	2.0	39,505	-	14,699					
2	Armament doors	-	-	-	-	-	-	Closed	Max.	Max.
8	Armament doors	-	-	-	-	-	-	10°	Max.	Max.
13_C	Armament doors	-	-	-	-	-	-	Opened	Max.	Max.
14_C	Armament doors	-	-	-	-	-	-	Opened	Max.	Max.
	Speed brakes	-	-	-	-	-	-	50°	Max.	
3204	Radome and forward, Fuselage Station 102, accelerated pushover; pitching acceleration ($\ddot{\theta}$)+2.474; altitude=0	-2.3	33,119	-	-	-	-	-	-	-

CRITICAL AREA: Radome and forward fuselage--max. negative shear and bending.

* B denotes F-106B; all other conditions pertain to the F-106A.

TABLE 8
MISCELLANEOUS COMPONENTS TESTS (F-106*)

TEST	CONDITIONS	POSITION	LOAD (LBS)
4B*	Rudder, C.P. --W.L. 90.6 Fuselage Station 7044		6118
1	Ejection seat and support rails**, lap belt at 45° to seat bottom	Forward crash	5760, lap belt 3600, forward shoulder harness
2	Ejection seat and support rails, all loads rotated 20° to side	Side crash	5760, lap belt 3600, forward shoulder harness
3	Ejection seat and support rails, load reacted at catapult tube	Catapult	9800, seat bottom
5D	Engine mount	Forward mount, left hangar	24,042 down
2F	Engine mount	Forward mount, left hangar	17,328 up
19F	Engine mount	Forward mount trunnion	13,489 outbd 40,013 forward 3105 up
5E	Engine mount	Forward mount trunnion	2838 outboard 37,151 forward 28,721 down
5C	Engine mount	Forward mount trunnion	9739 inboard 16,662 aft 18,523 down
Emergency Landing	Engine mount	Forward mount trunnion	9,200 outbd 52,640 forward 9,920 down
1910C	Engine mount	Aft mount hangar	20,024 compression
1804C	Engine mount	Aft mount hangar	39,903 tension

* B denotes F-106B; all other conditions pertain to the F-106A.
** The ability of the seat to support ejection air loads is confirmed by rocket sled tests in lieu of the less realistic static test loads.

TABLE 9

MISCELLANEOUS COMPONENTS TESTS (F-106**)

	TEST CONDITION	OPEN	CLOSED	AIR	DRAG LOADS	INERTIA LOADS
	Drag chute attach fitting, applied to longitudinal axis at +18° and at -5°.				14, 400 ultimate*	
1c	Ram air turbine support structure		x	Max. opening		Max.
3	Ram air turbine support structure	22°		Max. opening		Max.
1	Wing doors (main landing gear)		locked			
6	Wing doors (main landing gear)	20°				
7B	Wing doors (main landing gear)	locked				
1	Fuselage doors (main landing gear)		locked			
3	Fuselage doors (main landing gear)	locked				
	Door (nose landing gear)		locked			
	Door (nose landing gear)	full				

* Ultimate loads are compression (-), tension (+).

** B denotes F-106B; all other conditions pertain to the F-106A.

TABLE 10

COMPONENTS TESTS (F-106A)
(CONTROL SYSTEMS)

	TEST CONDITION	POSITION	LO/ (LBS) UL
7	Rudder, load reacted at servo valve stops	Neutral	450
8	Rudder, load applied to lefthand pedal and reacted at servo valve stops	Full right	450
9	Rudder, load applied to lefthand pedal and reacted at rudder system stops	Full left	450
	Rudder, feel system, pressure supplied to cylinder, pilot effort loads applied to pedal		2250 ps ultimat
2	Elevator, control	Elevator full up, stick forward	350 ard
3	Elevator, surfaces	Elevator full down, stick aft, pressure off	350 t,
4	Elevator, actuators,	Elevator full up, stick aft, pressure on	350
5	Elevator, surfaces	Elevator full down, stick forward	350 rwar
1	Elevator (feel system)	Elevator neutral	15 psi (
3a	Elevator, trim jack (feel system)	Full down, stick aft	15 psi (balance stick)
3b	Elevator, trim (feel system)	Full up, stick forward	15 psi (balance stick)
2	Aileron	Extreme right aileron, stick left	150, lef pressu
3	Aileron, load reacted at servo valve stops	Extreme left aileron, stick right	150, righ pressu
4	Aileron, system, load reacted by system stops	Extreme travel	150
1	Throttle (power system), load reacted at fuel control (Fuselage Station 526.25) or at lever quadrant stops	Off	75
2	Throttle (power system), load reacted at fuel control (Fuselage Station 526.25) or at lever quadrant stops	Half on	75
3	Throttle (power system), load reacted at fuel control (Fuselage Station 526.25) or at lever quadrant stops	Full on	75
	Brake (pedal toes), load applied to each toe simultaneously	Mid-adjust of rudder bars	450

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TABLE 10

COMPONENTS TESTS (F-106A)
(CONTROL SYSTEMS)

TEST CONDITION	POSITION	LO/ (LBS) UL	LOAD (LBS) ULTIMATE
der, load reacted at /o valve stops	Neutral	450	450
der, load applied to lefthand al and reacted at servo valve stops	Full right	450	450
der, load applied to lefthand pedal reacted at rudder system stops	Full left	450	450
der, feel system, pressure supplied ylinder, pilot effort loads applied to al		2250 ps ultimat	2250 psi pressure ultimate*
vator, control	Elevator full up, stick forward	350 ard	350
vator, surfaces	Elevator full down, stick aft, pressure off	350 t,	350
vator, actuators,	Elevator full up, stick aft, pressure on	350	350
vator, surfaces	Elevator full down, stick forward	350 rward	350
vator (feel system)	Elevator neutral	15 psi (	15 psi (cylinder)
vator, trim jack (feel system)	Full down, stick aft	15 psi (balance stick)	15 psi (cylinder, balances load on stick)
vator, trim (feel system)	Full up, stick forward	15 psi (balance stick)	15 psi (cylinder, balances load on stick)
eron	Extreme right aileron, stick left	150, left pressu	150, no actuator pressure
eron, load reacted at servo ve stops	Extreme left aileron, stick right	150, right pressu	150, no actuator pressure
eron, system, load reacted system stops	Extreme travel	150	150
rottle (power system), load icted at fuel control (Fuselage tion 526.25) or at lever quadrant ps	Off	75	75
rottle (power system), load icted at fuel control (Fuselage tion 526.25) or at lever quadrant ps	Half on	75	75
rottle (power system), load acted at fuel control (Fuselage ition 526.25) or at lever quadrant ps	Full on	75	75
ake (pedal toes), load applied each toe simultaneously	Mid-adjust of rudder bars	450	450

2

TABLE 11

COMPONENTS TESTS
(PYLON AND TANK*)

TEST CONDITIONS		n_y	n_z	Altitude	Mach	Flight Pattern	Critical Areas
1504	Tanks, empty;		-3.0	Sea level	.95	Symmetric pushover	Max.vertical load in pylon aft support post.
8	Tanks, empty; max. pitching acceleration	$(\ddot{\xi})$		Sea level	.90	Level flight roll	Max.pylon chock load.
9	Tanks, empty;	max.		Sea level	.95	Og roll	Max.tank side shear and moment.
12	Tanks, full	max.		15,000	.52	Bank to bank roll	Max.pylon forward pad load, max. pylon hook load, max. aft tank side and vertical shear.

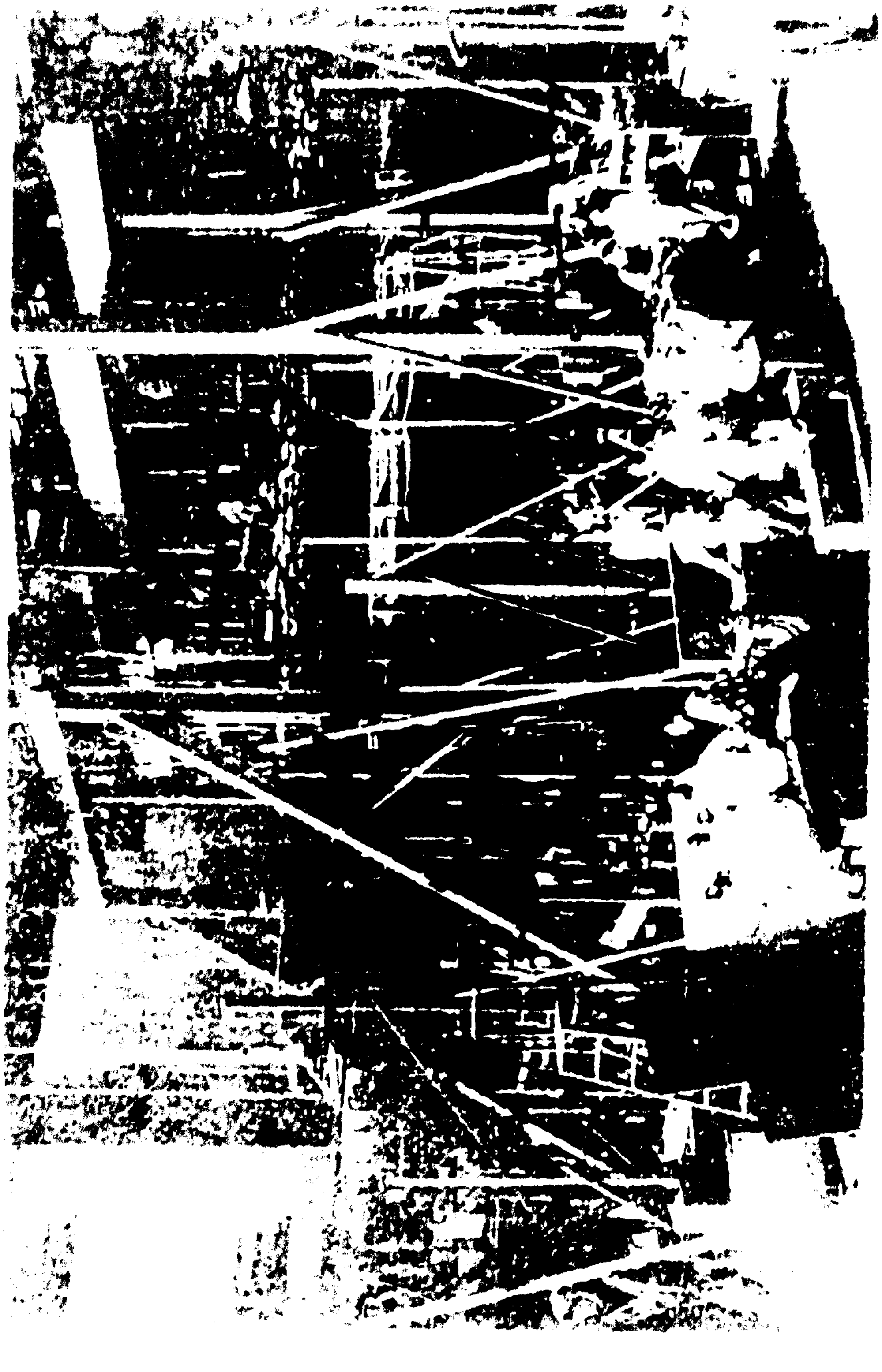
* Complete tank tests also conducted by Royal Jet Company.

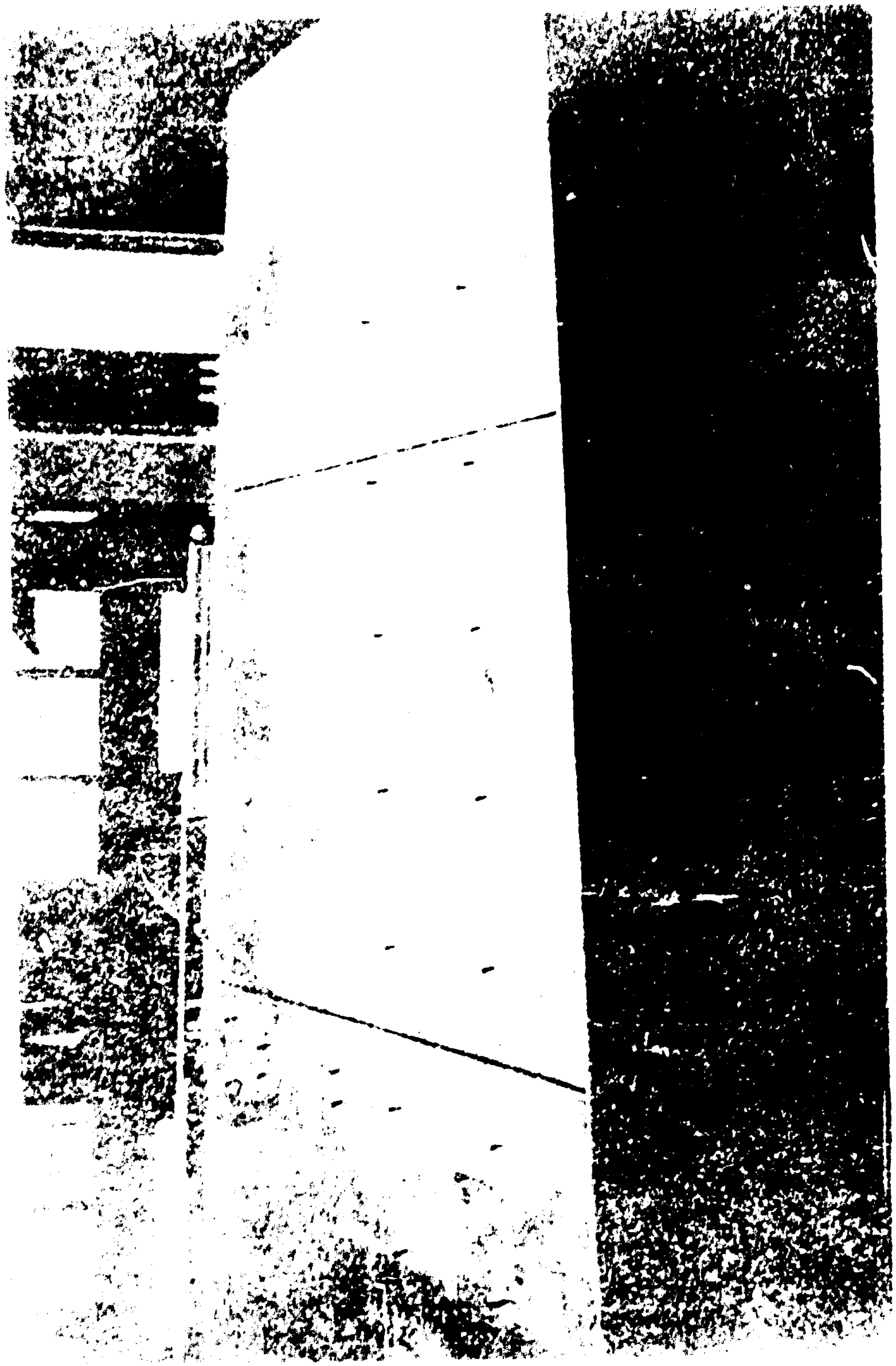
TABLE 12

COMPONENTS TESTS
(VARIABLE RAMP)

TEST CONDITIONS		Altitude	Mach	Axial loads (lbs) (ultimate)*	Critical Areas
7	Ramp at 30° angle; control failure	Sea level	1.138	-	High positive pressures on the forward and aft ramps.
8	Ramp at 17° angle; mild buzz	30,000	1.9	-	Max. negative pressures on forward ramp.
11(+)	Ramp at 17° angle; severe buzz		1.9	-	Max. overall positive pressures on all ramps.
11(-)	Ramp at 17° angle; severe buzz		1.9	-	Max. overall negative pressures on all ramps.
VARIABLE RAMP ACTUATOR					
7	Actuators, aft Actuators, forward			-14,888 + 2,830	
8	Actuators, aft Actuators, forward			- 867 + 4,133	
11(+)	Actuators, aft Actuators, forward			-12,399 - 404	
11(-)	Actuators, aft Actuators, forward			+15,255 + 6,455	

* Ultimate loads are compression (-), tension (+).





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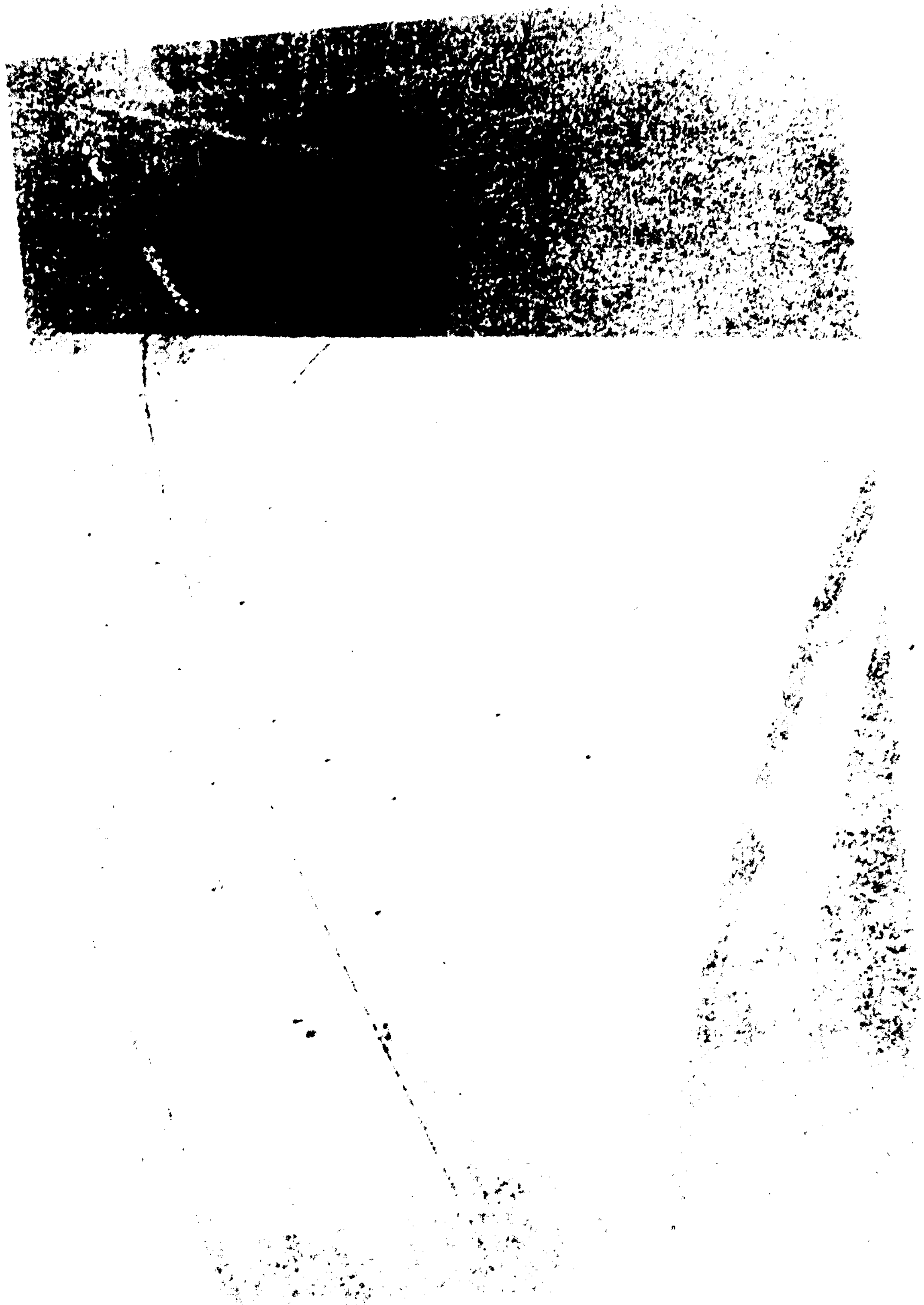

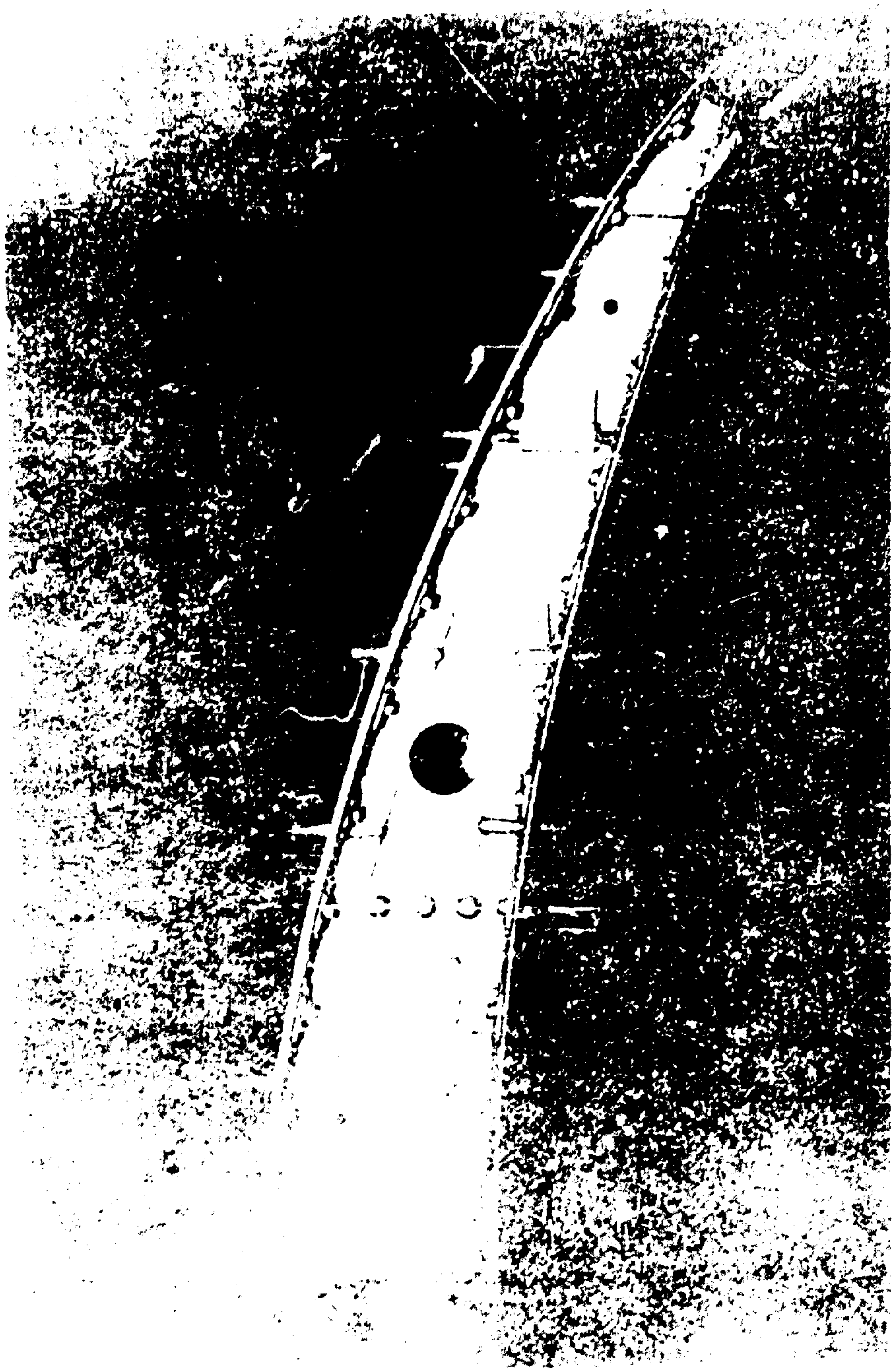

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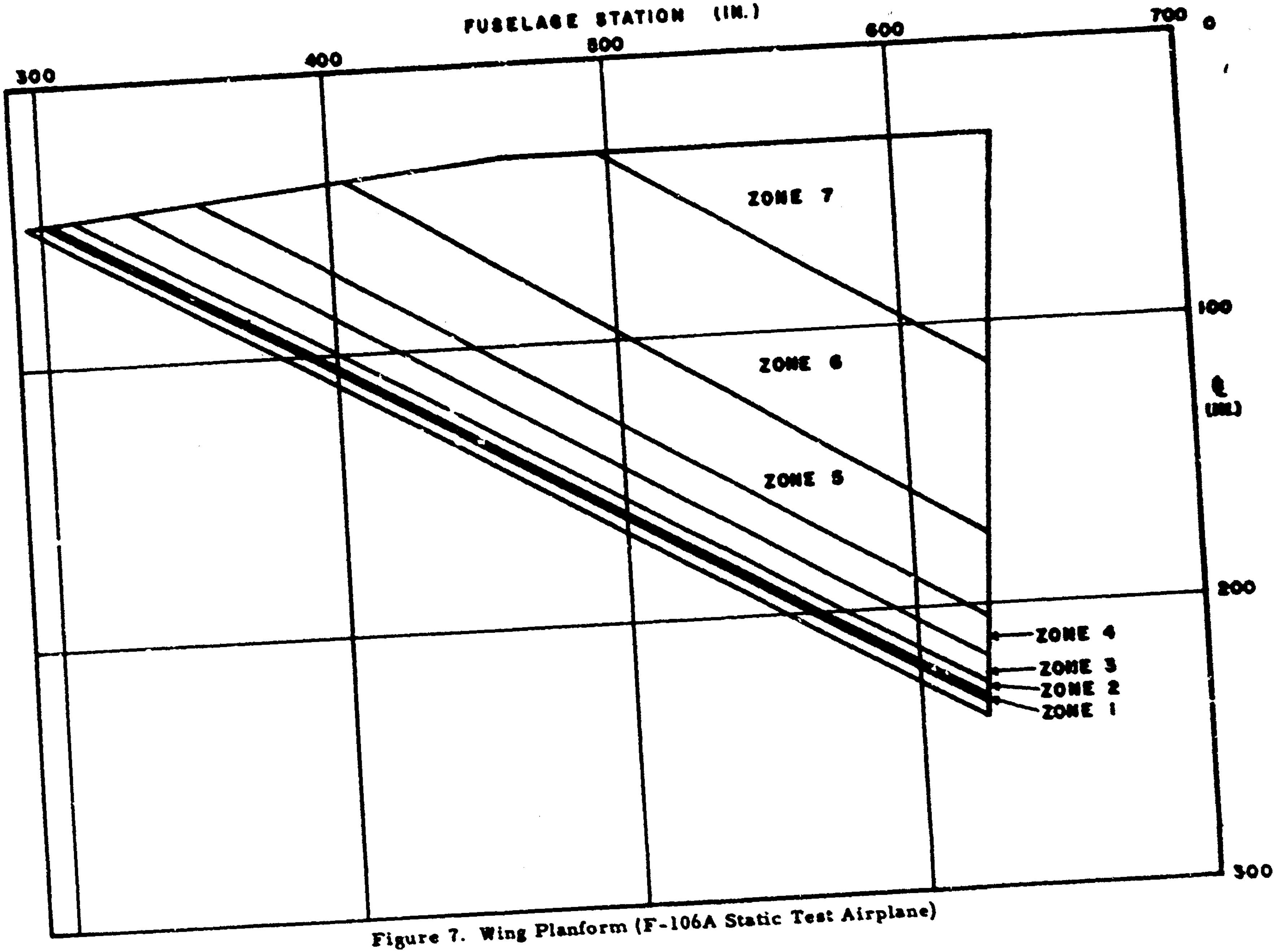

Figure 7. Wing Planform (F-106A Static Test Airplane)

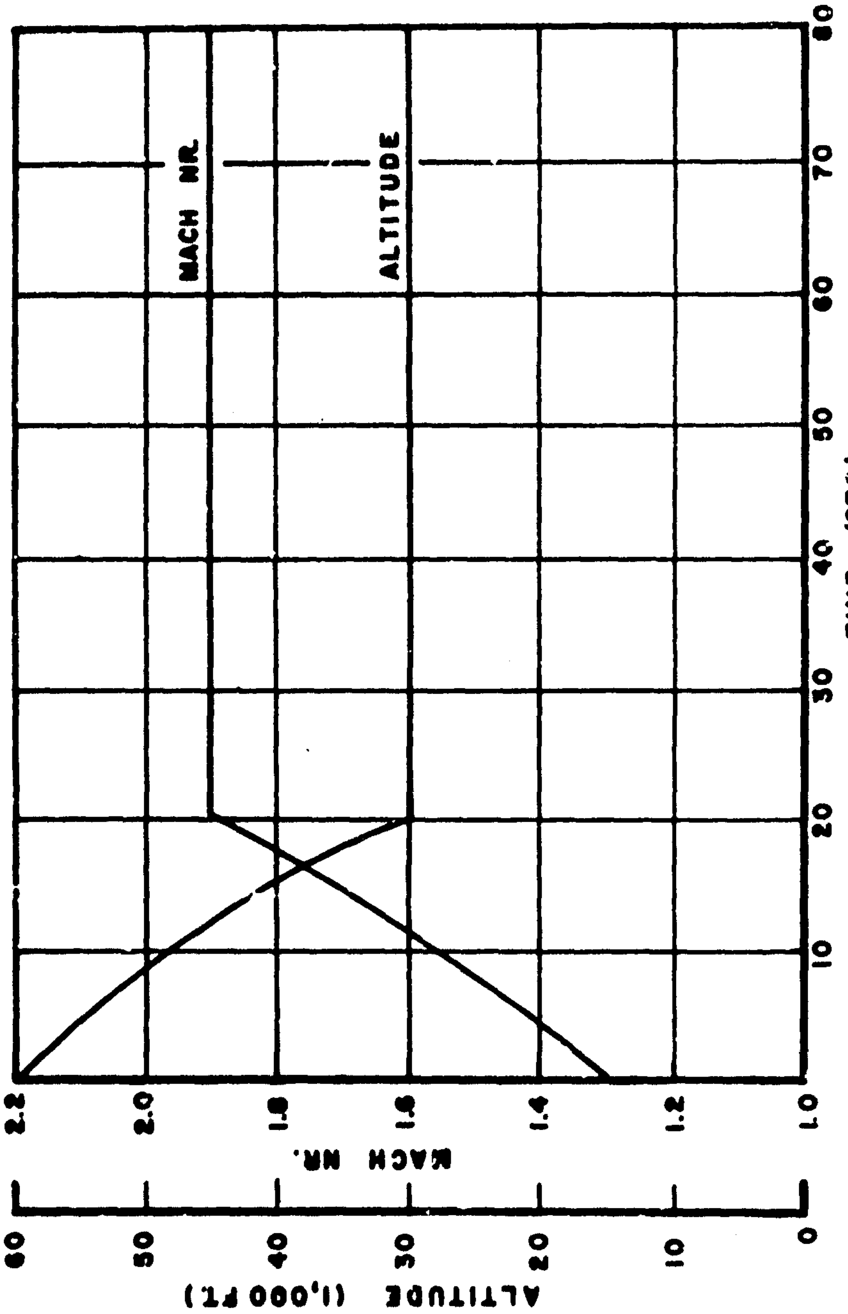

Figure 8a. Mach Nr. and Altitude Vs Time (F-106A Static Test Airplane)

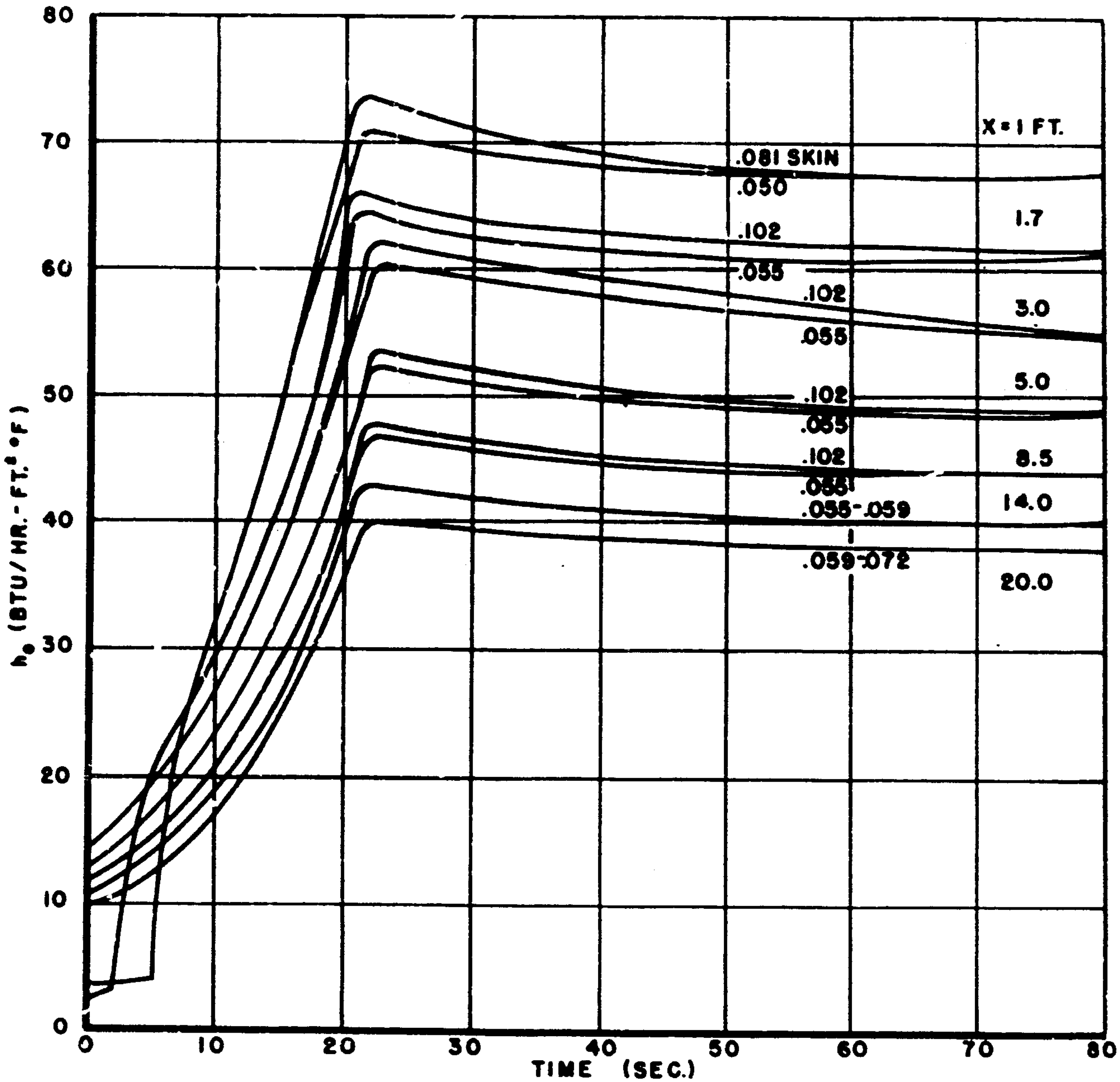

Figure 8b. Heat Transfer Coefficients Vs Time (F-106A Static Test Airplane, 60° Power Dive)

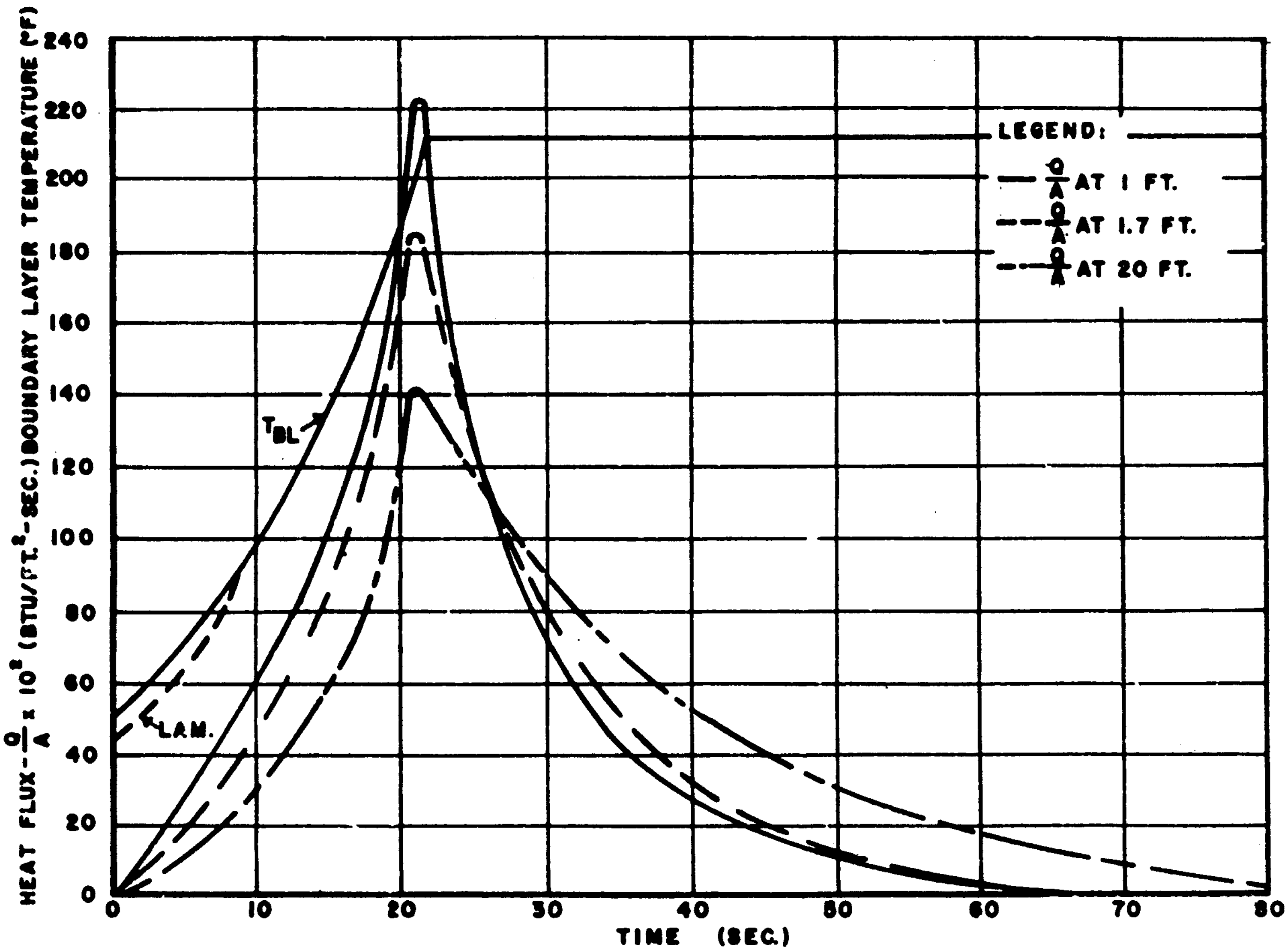

Figure 8c. Heat Flux and Boundary Layer Temperature Vs Time (F-106A Static Test Airplane, 60° Power Dive)

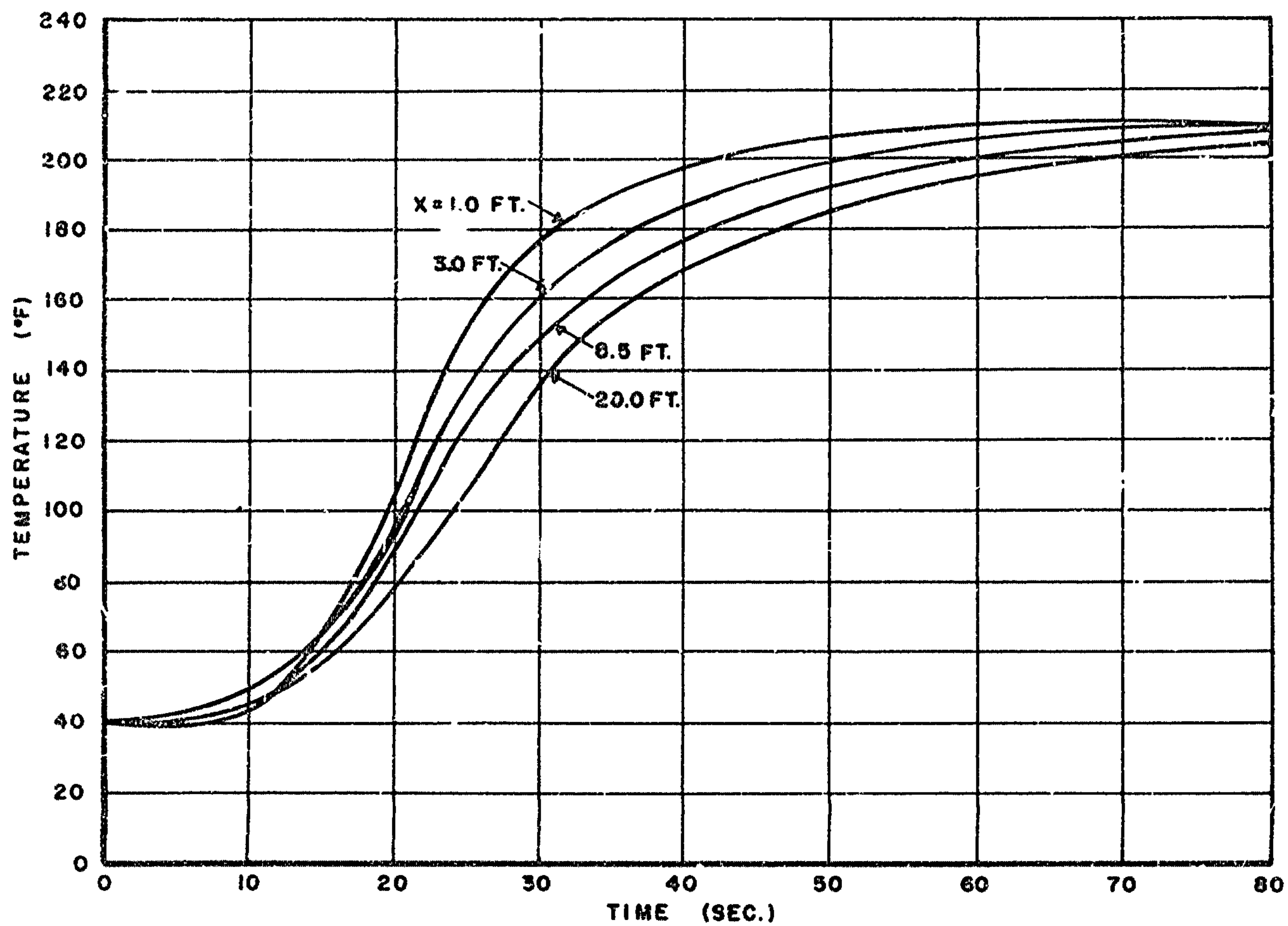

Figure 8d. Skin Temperature Vs Time (F-106A Static Test Airplane with .055 Aluminum Skin, 60° Power Dive)

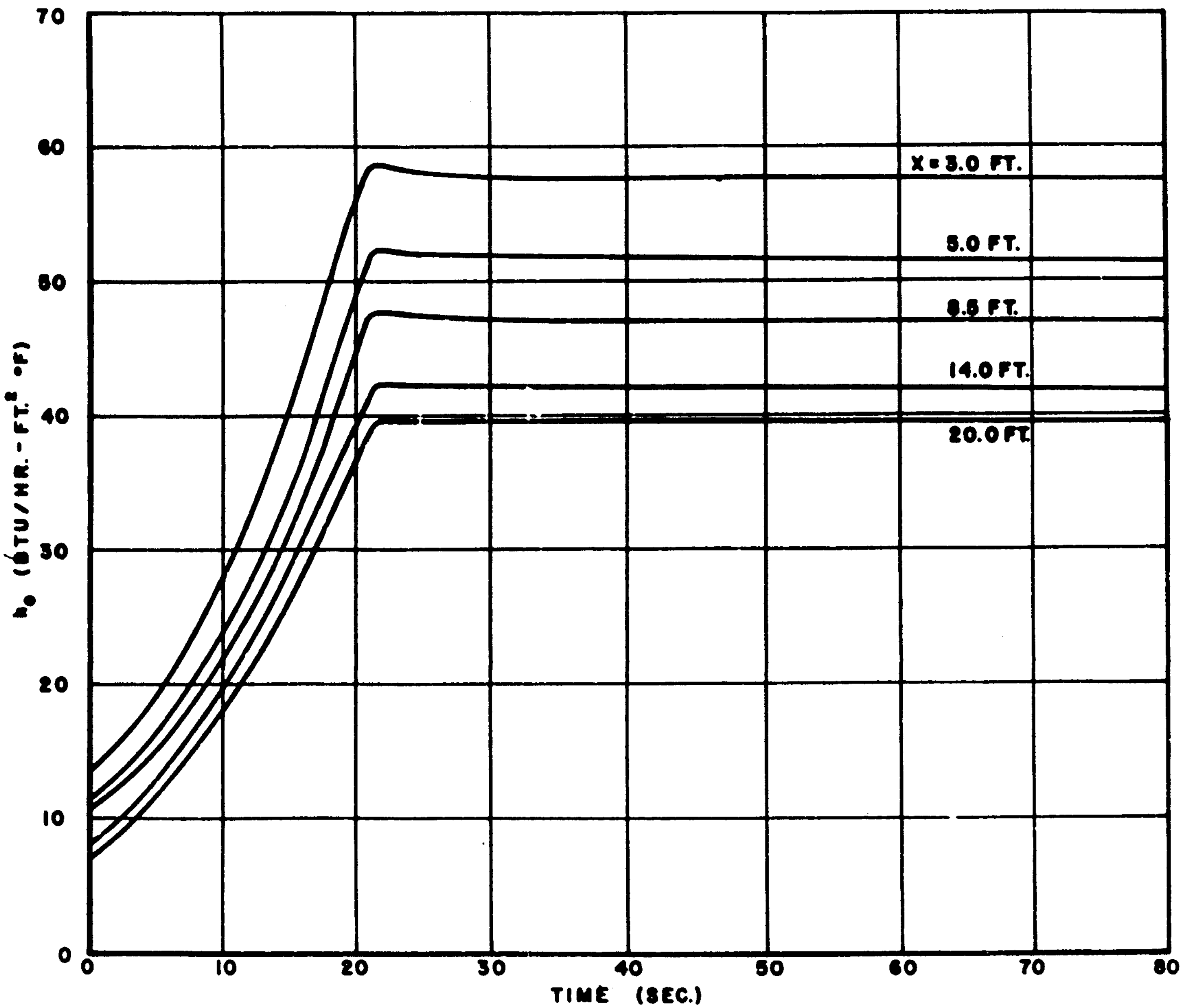

Figure 8e. Heat Transfer Coefficients Vs Time (F-106A Static Test Airplane with Fuel, 60° Power Dive)

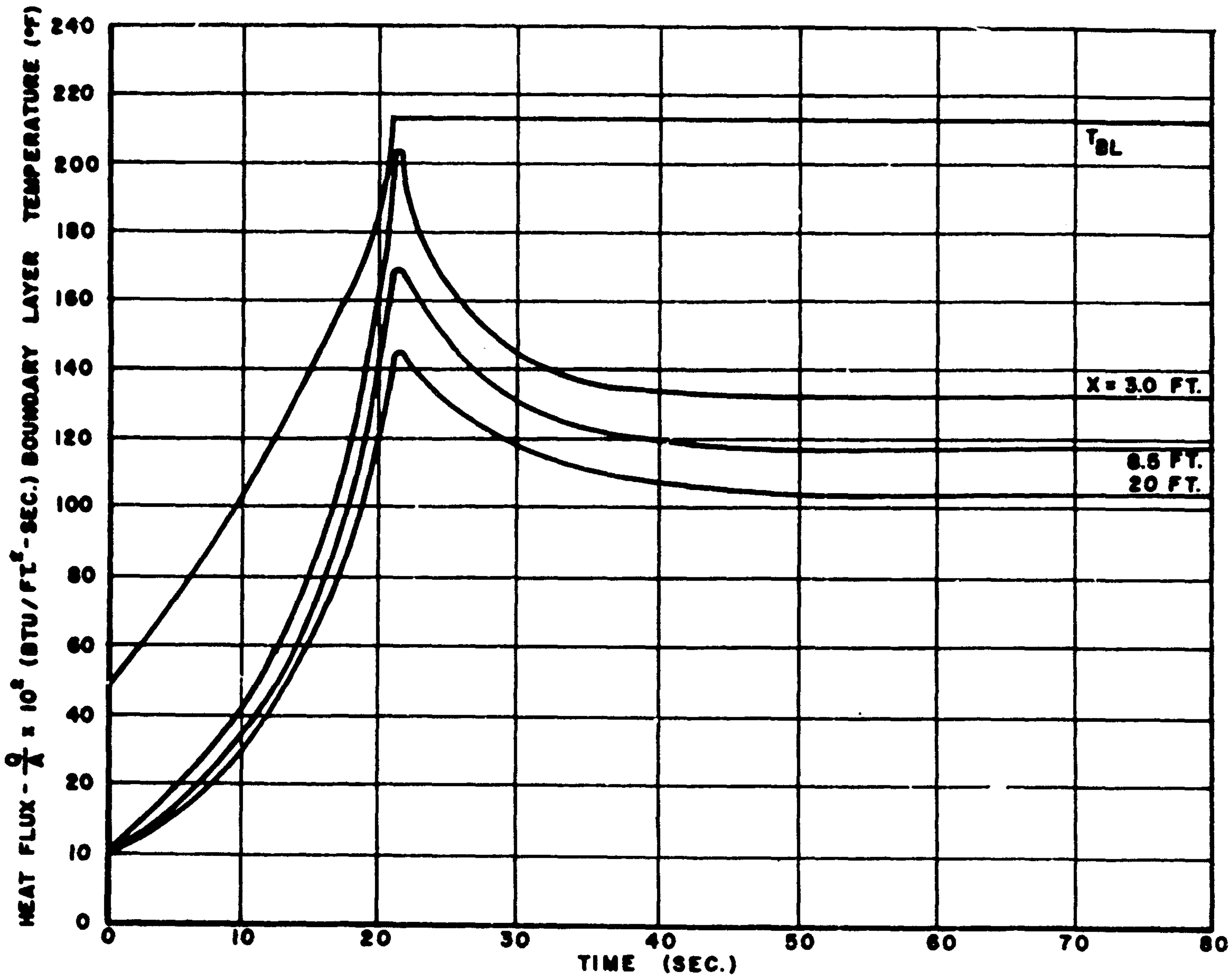

Figure 8f. Heat Flux and Boundary Layer Temperature Vs Time (F-106A Static Test Airplane with Fuel, 60° Power Dive)

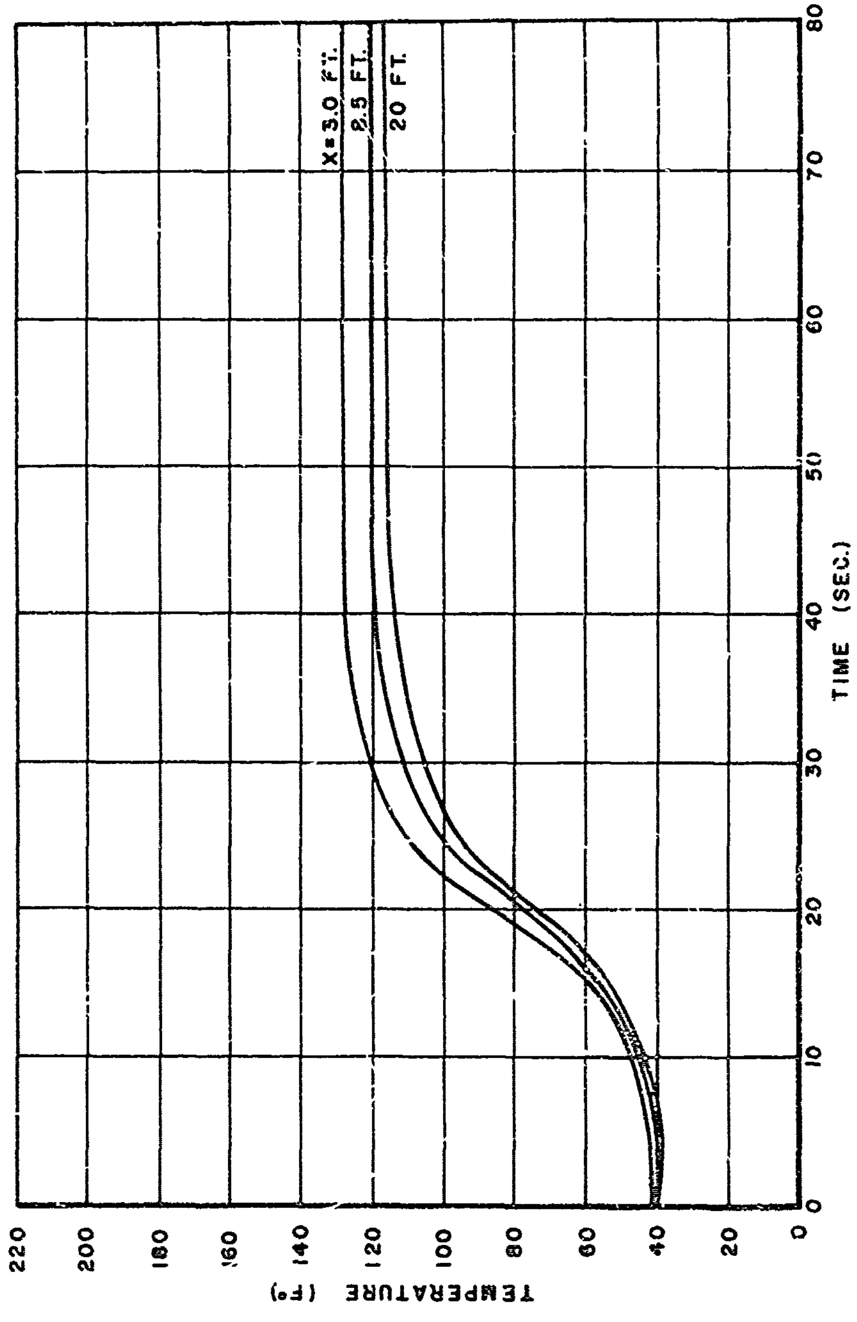

Figure 8g Skin Temperature Vs Time (F-106A Static Test Airplane with Fuel, 60° Power Dive)

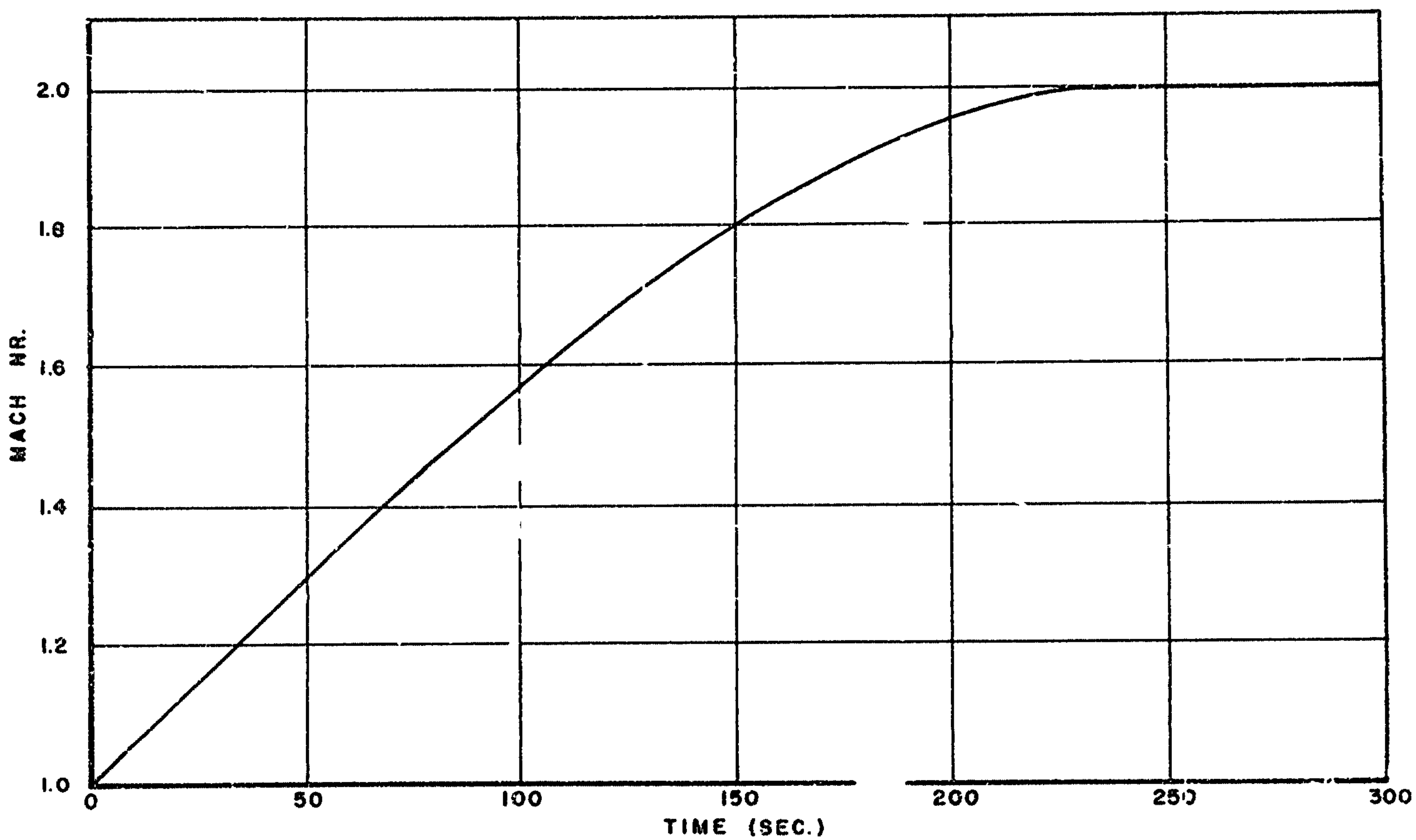

Figure 9a. Mach Nr. Vs Time at 35,000 Ft Altitude (F-106A Static Test Airplane in Level Flight with Acceleration to Mach 2.0)

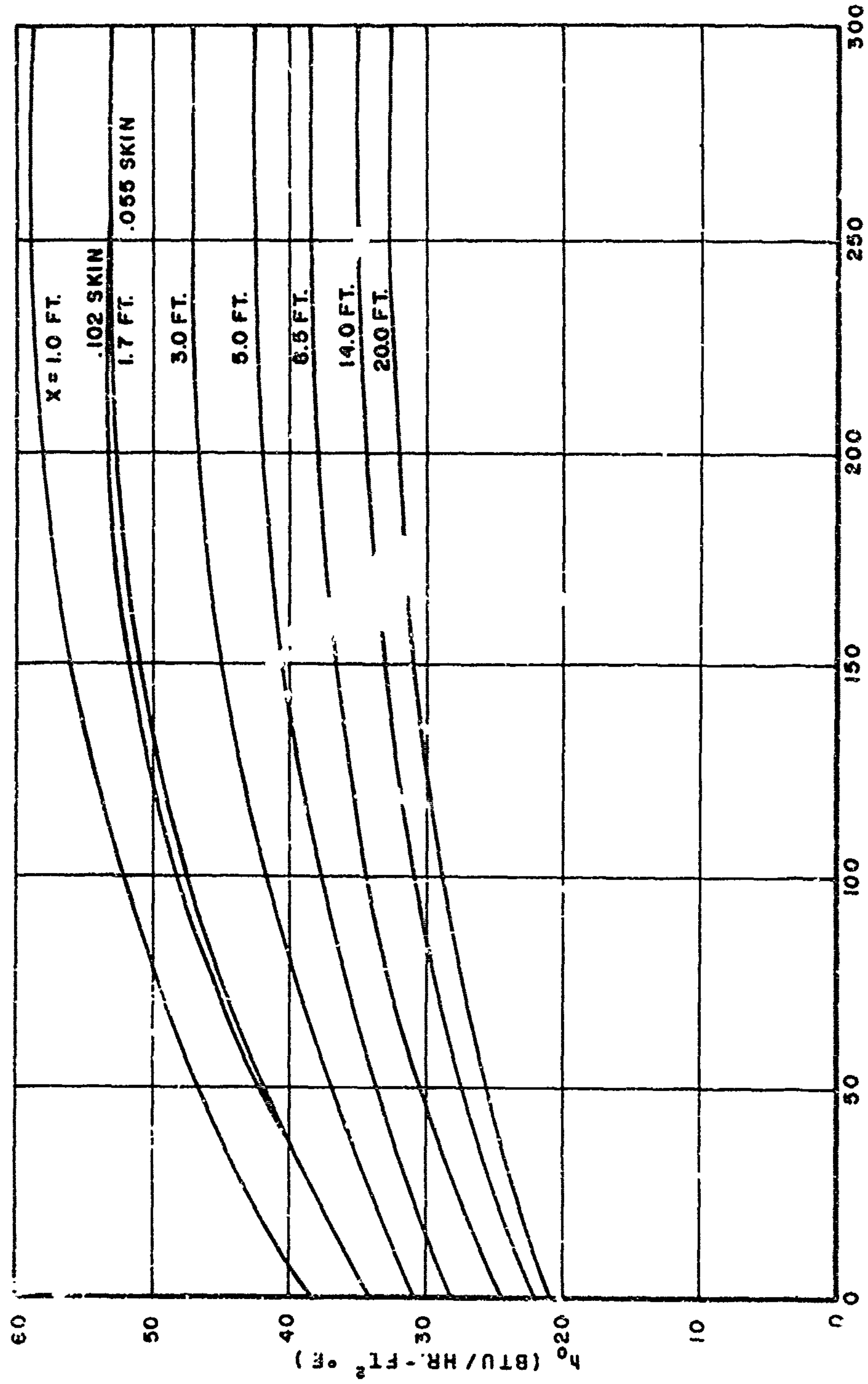

Figure 9b. Heat Transfer Coefficient Vs Time (F-106A Static Test Airplane with Acceleration of Mach 2.0)

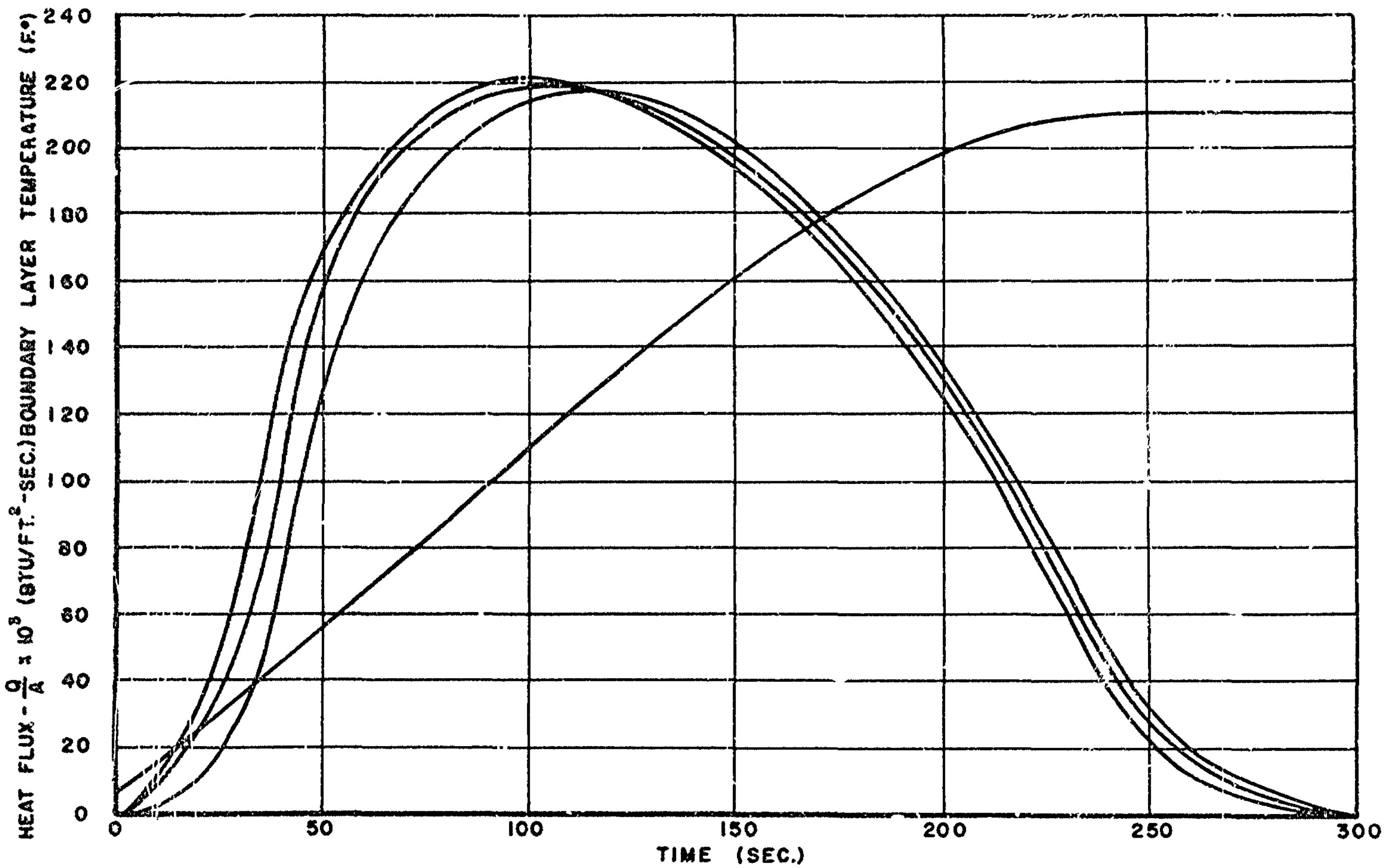

Figure 9c. Heat Flux and Boundary Layer Temperature Vs Time (F-106A Static Test Airplane with .055 Aluminum Skin, and an Acceleration of Mach 2.0)

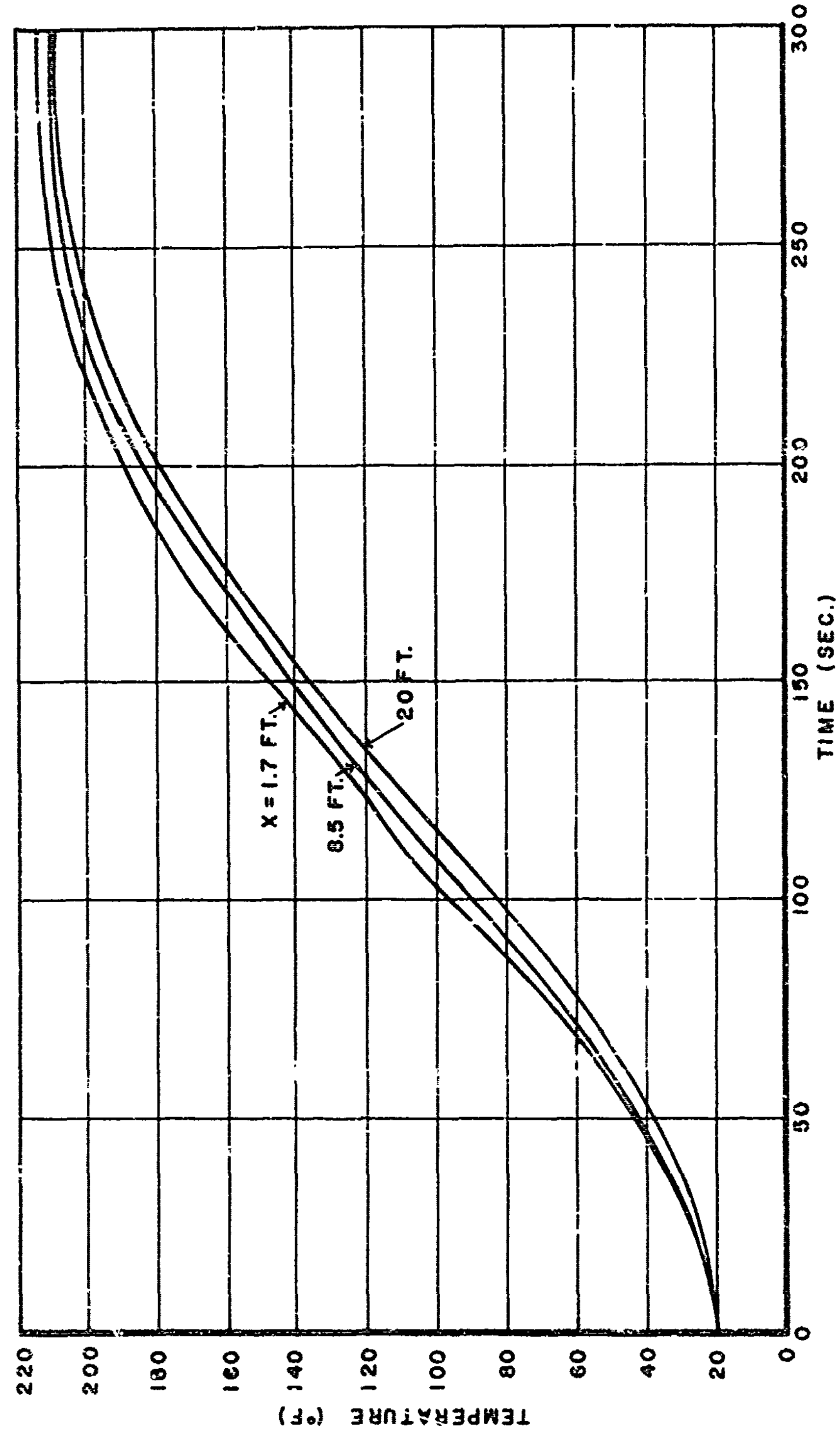

Figure 9d. Skin Temperature Vs Time (F-106A Static Test Airplane with .055 Aluminum Skin and an Acceleration of Mach 2.0)

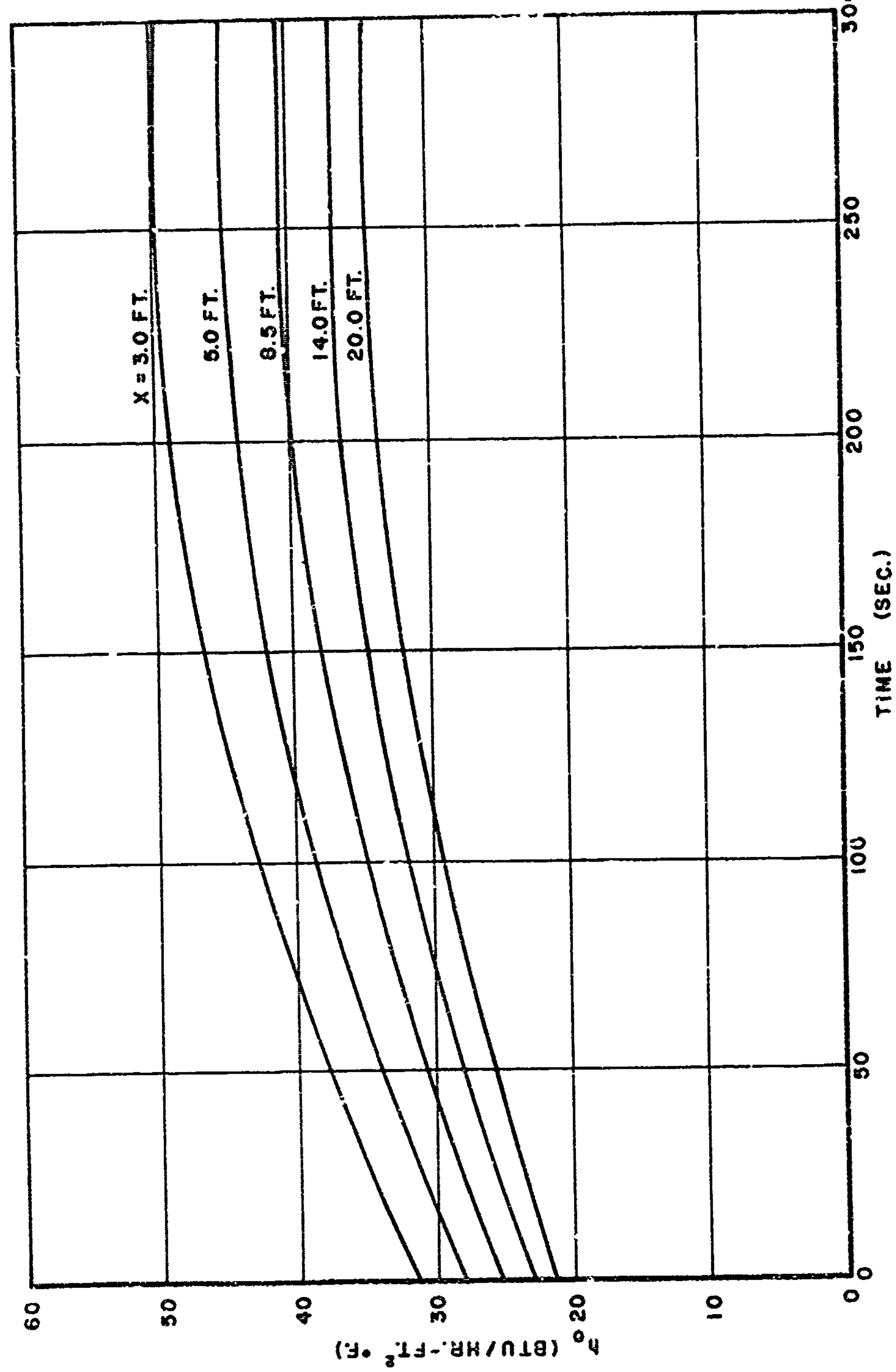

Figure 9e. Heat Transfer Coefficients Vs Time (F-106A Static Test Airplane with Fuel and with an Acceleration of Mach 2.0)

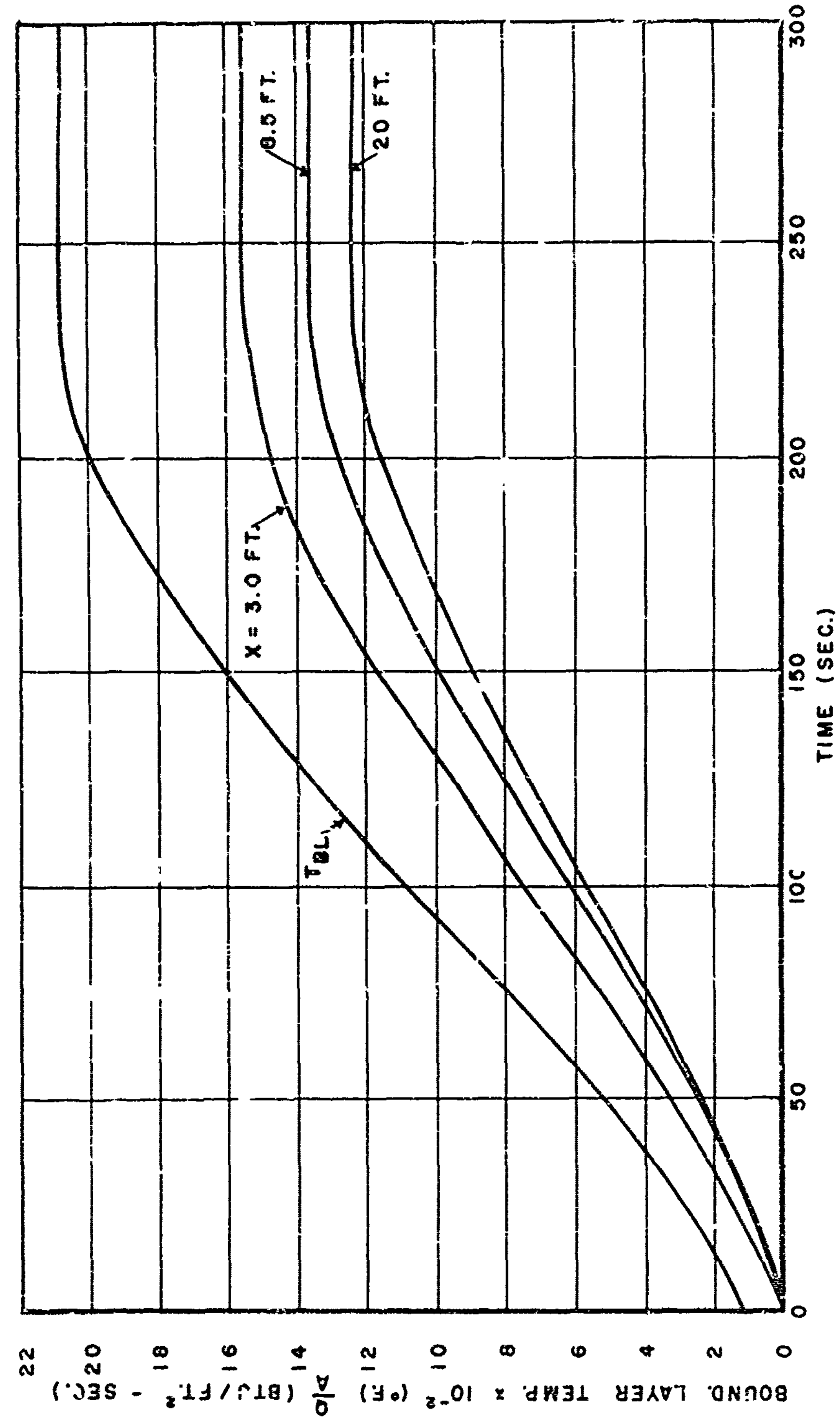

Figure 91. Heat Flux and Boundary Layer Temperature (F-106A Static Test Airplane of .055 Aluminum Skin, Operating with Fuel and at an Acceleration of Mach 2.0)

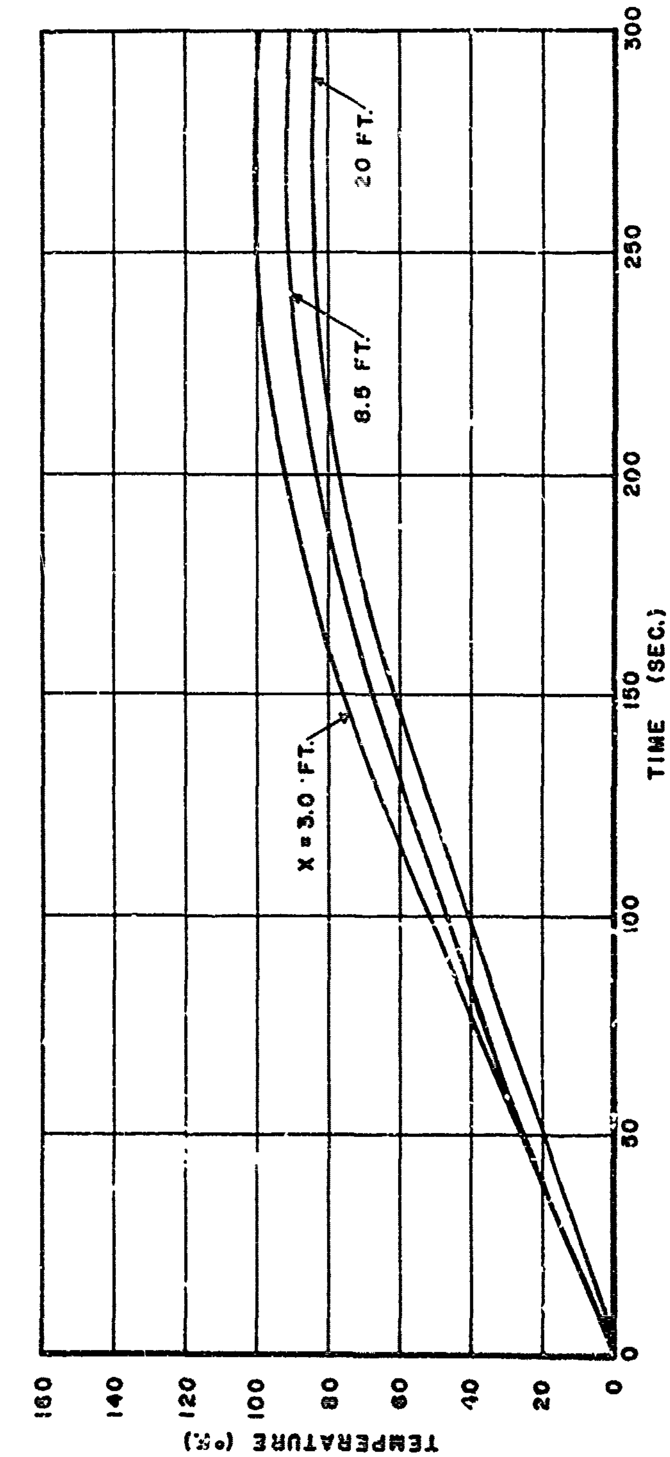

Figure 9g. Skin Temperature Vs Time (F-106A Static Test Airplane of .055 Aluminum Skin, Operating with Fuel and at an Acceleration of Mach 2.0)

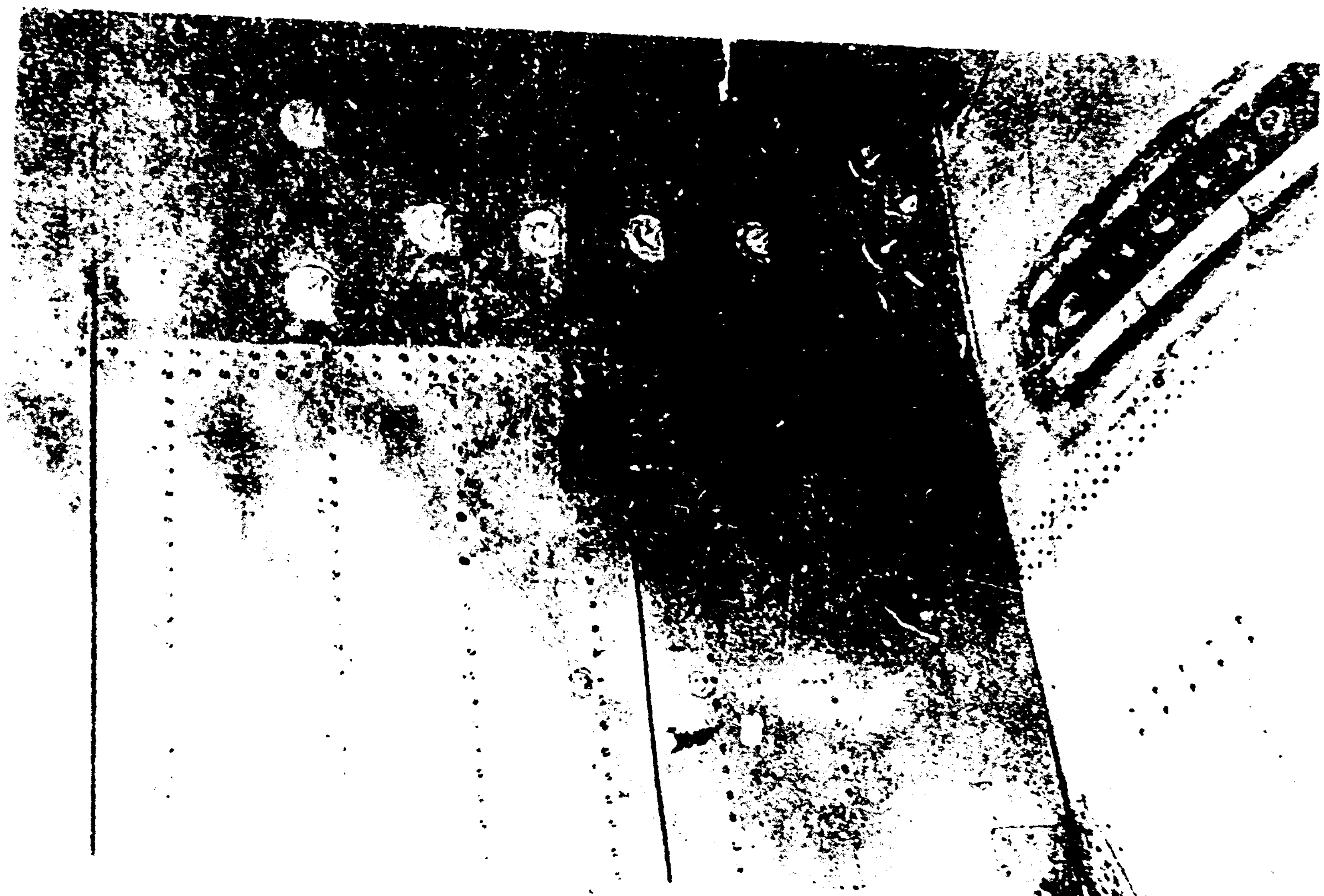

Figure 10. Inboard Right Hand Elevon at 67 Percent Design Ultimate Load (F-106A Test Condition 1704)

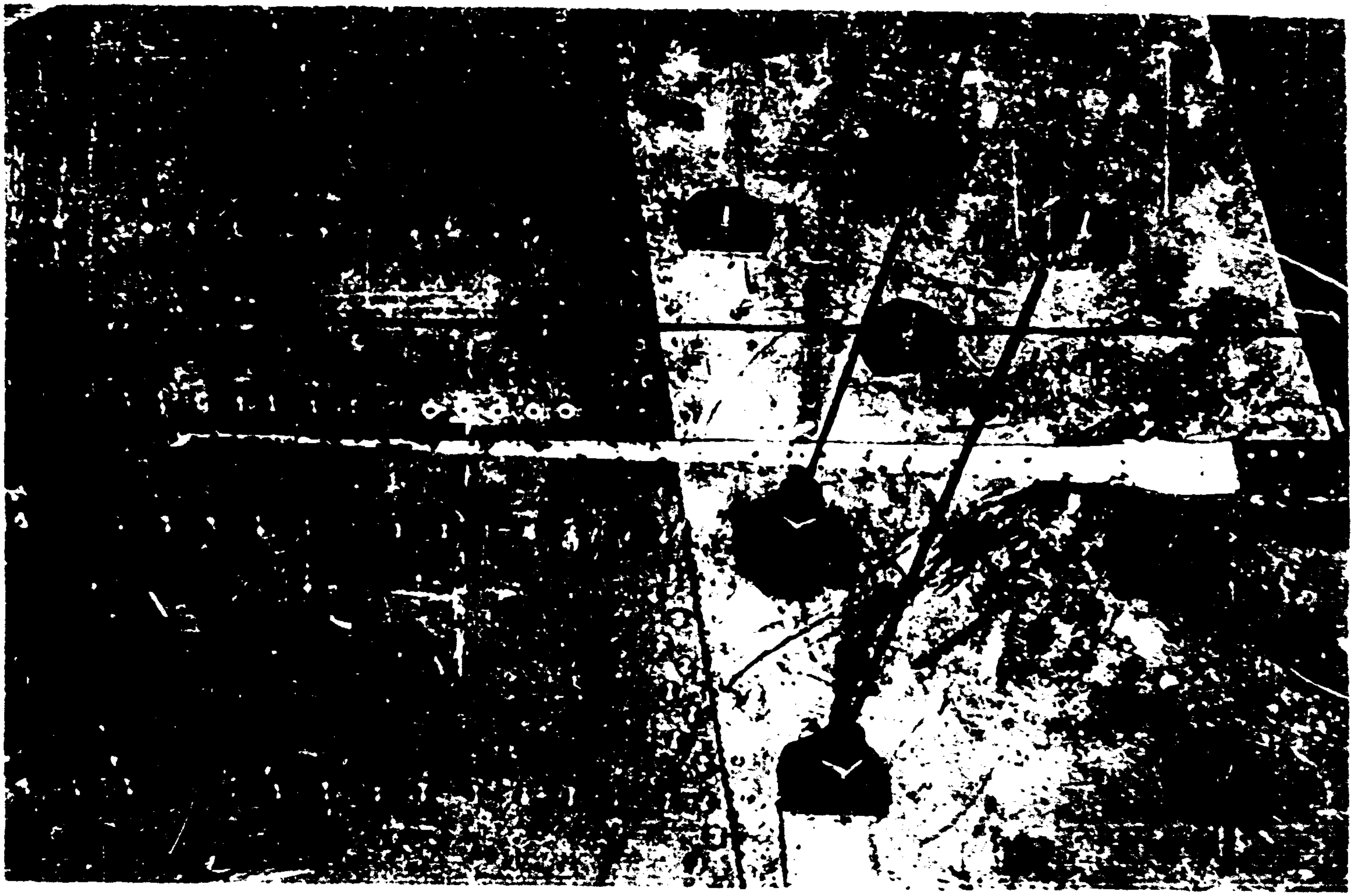

Figure 11. Elevon Lap Joint Damage Due to Separation and Jamming After 100 Percent Ultimate Load (F-106A Increased Fuel Test Condition 2502)

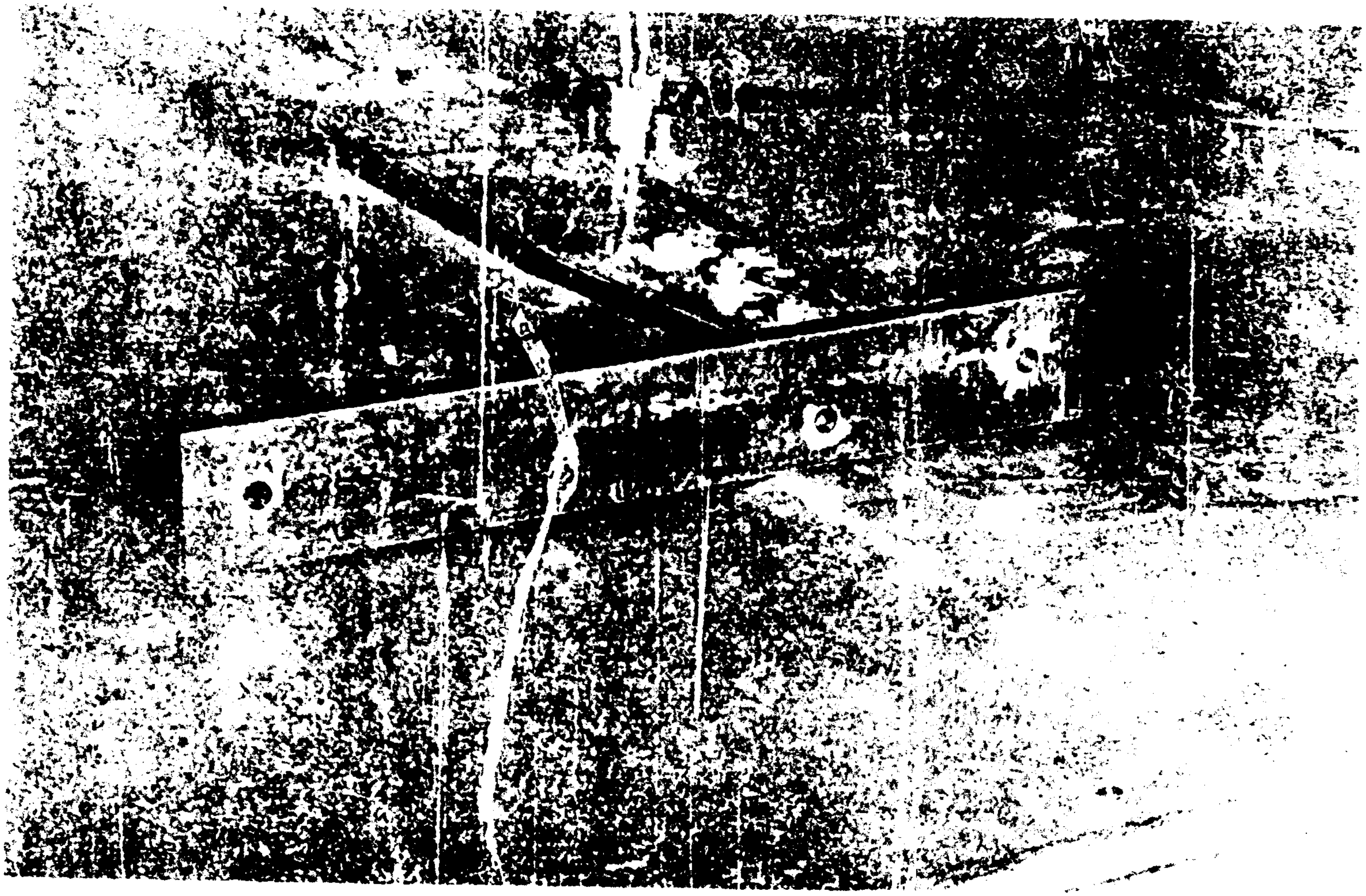

Figure 12. Wing Upper Skin Compression Failure Just Inboard of B. L. 99.94 After Failure (F-106A Increased Fuel Test Condition 3202 at 95 Percent Ultimate Load)

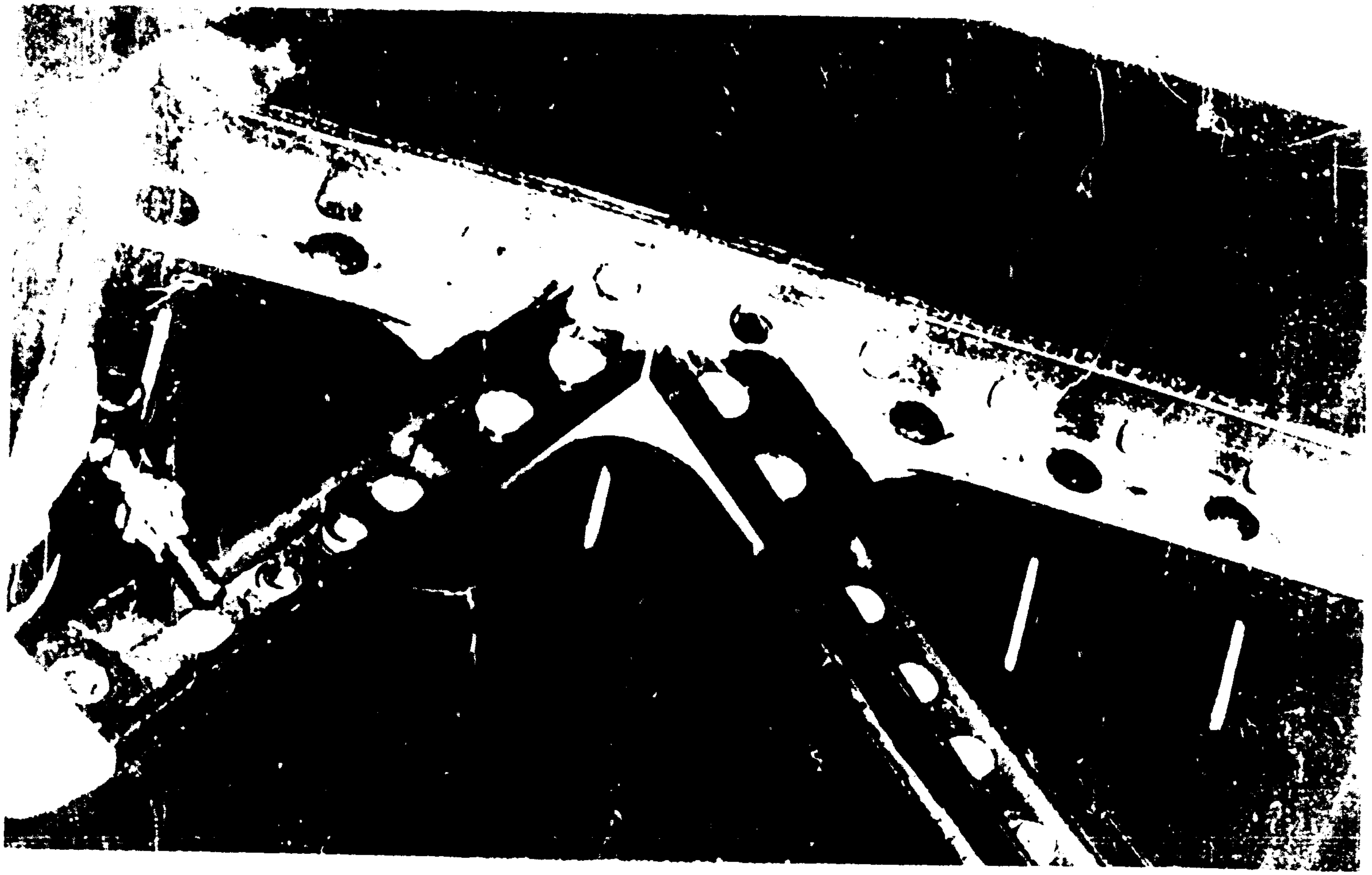

Figure 1[illegible]. Failed Rib Cap Web at B. L. 92.94 Rib After Failure, Showing Crack Between Holes (F-106A Increased Flap) Test Condition 3202 at 95 Percent Ultimate Load

Figure 14a. Right Main Landing Gear Forward Drag Brace After Compression Failure, Showing Gear Failure (F-106B Increased Fuel Test Condition 1102 B at 135 Percent Design Ultimate Load)

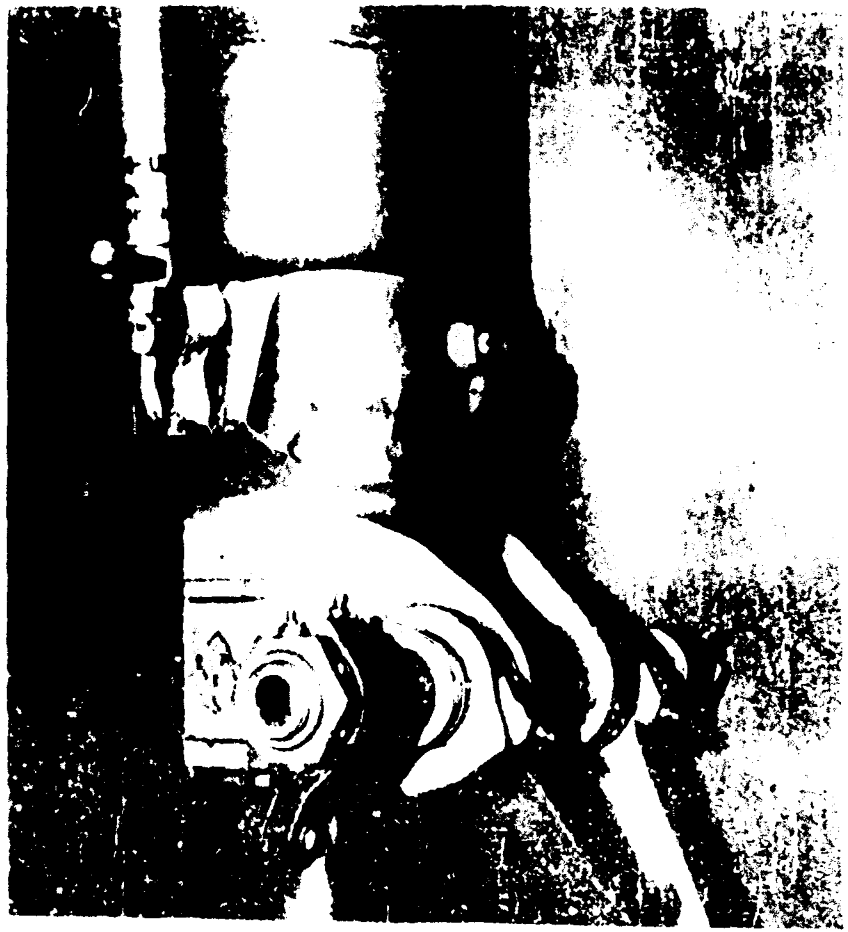

Figure 14b. Right Main Landing Gear Forward Drag Strut Attachment Lug After Gear Failure (F-106B Increased Fuel Test Condition 1102 B at 135 Percent Design Ultimate Load)

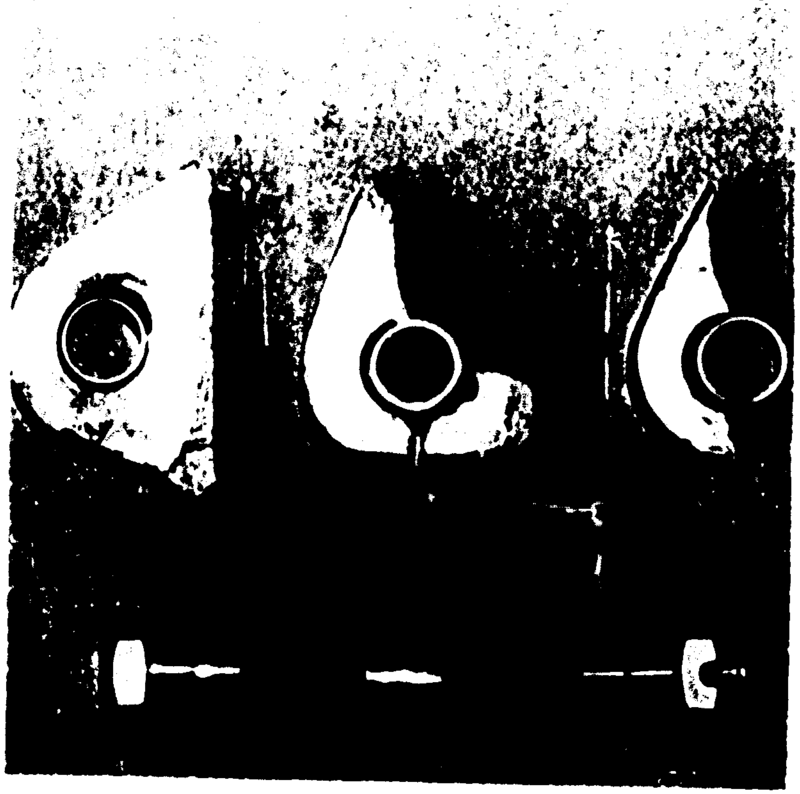

Figure 14c. Right Main Landing Gear Forward Drag Brace Attachment Lug and Bolt After Failure (F-106B Increased Fuel Test Condition 1102 B at 135 Percent Design Ultimate Load)

Figure 14d. Right Main Landing Gear Aft Drag Brace Upper Attachment Bolt After Failure (F-106B Increased Fuel Test Condition 1102 B at 135 Percent Design Ultimate Load)

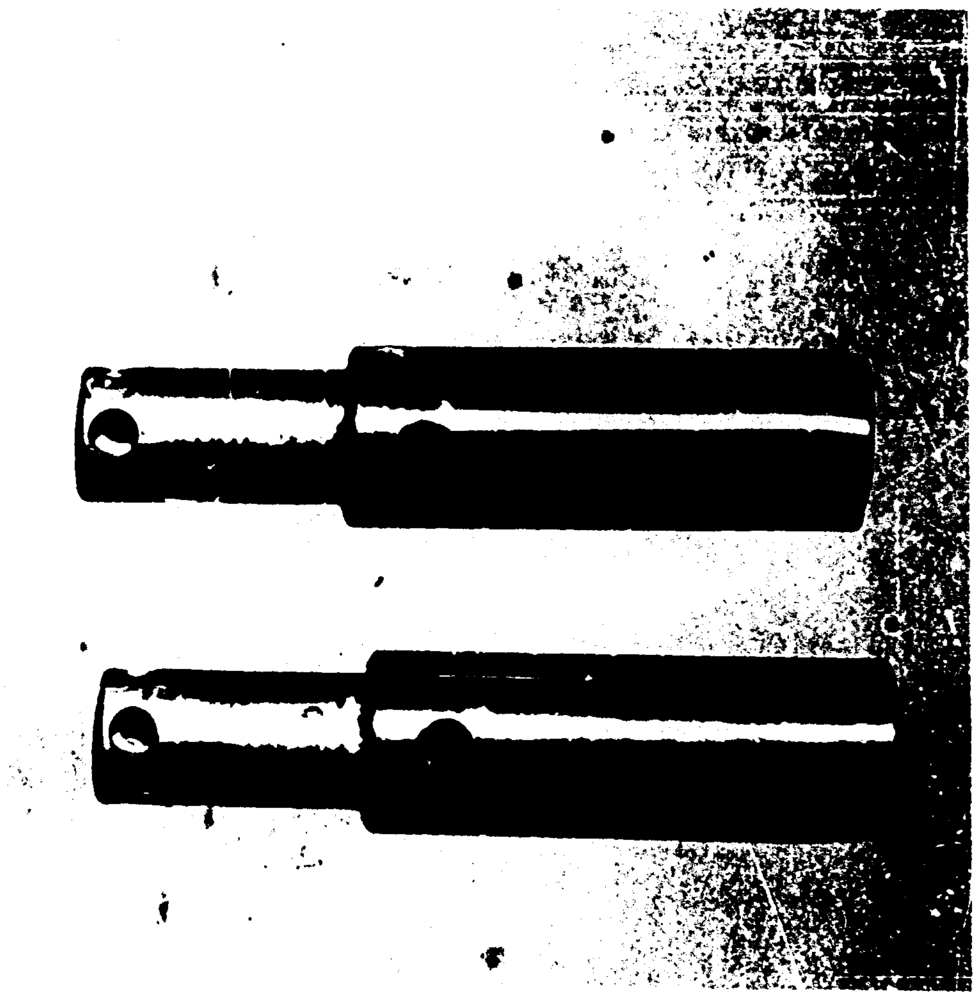

Figure 14e. Right Main Landing Gear Trunnions After Failure (F-106B Increased Fuel Test Condition 1102 B at 135 Percent Design Ultimate Load)